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FORT EUSTIS, VIRGINIA

TCREC TECHNICAL REPORT 62-22

AUTOMATIC CONTROL OF STATIC ELECTRICITY FOR ARMY HELICOPTERS

Task 9R38-01-017-30

Contract DA 44-177-TC-652

October 1962

prepared by:

CORNELL AERONAUTICAL LABORATORY, INC.,
Buffalo, New York

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Prepared by

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Project Engineer
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for

U. S. ARMY TRANSPORTATION RESEARCH COMMAND
Fort Eustis, Virginia

HEADQUARTERS
U. S. ARMY TRANSPORTATION RESEARCH COMMAND
TRANSPORTATION CORPS
Fort Eustis, Virginia

TCREC-ADS 9R38-01-017-30

SUBJECT: Task 9R38-01-017-30, Helicopter Static Electricity

TO: See Distribution List

1. The work described in this report was accomplished by Cornell Aeronautical Laboratory, Inc., for the U. S. Army Transportation Research Command. The report documents the design, fabrication, and testing of an automatic system for the dissipation of static electricity from helicopters.

2. The conclusions and recommendations made by the contractor are concurred in by this Command.

3. Additional efforts are being expended by this Command in the research of static-charge generation and dissipation. Investigations are being made into other dissipation methods, the charging rates, etc., experienced in various atmospheres and the effects of operating variables on the generation properties of helicopter blades. Reports on these projects will be distributed as they become available.

FOR THE COMMANDER:

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PREFACE

Preliminary design considerations, based on results of completed work under Contract DA-44-177-TC-544 and other applicable studies, have shown a forced corona discharge technique to be the most feasible method by which a net charge on a rotary wing aircraft can be reduced to zero.

The charge on the aircraft can be measured with adequate precision with a specially designed electrostatic field sensor properly positioned on the underside of the aircraft in an area of minimum turbulence relative to main rotor downwash. Auxiliary control circuits link the sensor to a power supply to form a closed loop system to control the aircraft charge about the zero reference level.

A system design has been completed, which includes the necessary equipment suitable for use with various Army rotary wing aircraft within their operating limitations. Extensive field tests to determine system operational characteristics relative to an H-37 aircraft in a hot, dry environment were concluded.

ACKNOWLEDGEMENT

The following Laboratory personnel were active participants during the performance of this program.

C. J. Borkowski	-	Mechanical design and electronic fabrication of system components. Active participation in the field test program.
R. A. Hayman	-	Design of control circuitry.
T. D. Mahany	-	Redesign of sensor electronics.
D. P. Springston	-	Instrumentation design along with coordination of high voltage power supply activities.
W. T. Harpster	-	Assisted with the editorial work.

The author wishes to acknowledge the excellent cooperation of personnel of the Army Aviation Test Office at Edwards Air Force Base, California and Aviation Section, 2nd Artillery Group, Ft. Niagara, N. Y. during flight test portions of the program.

Project Officer for this program was Mr. S. B. Poteate of the Transportation Research Command, Ft. Eustis, Va. His active interest and participation aided considerably in the performance of this program.

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I SUMMARY

Results of previous work, completed under Contract DA 44-177-TC-544, indicated a corona discharge technique to be the most practical method by which a net charge on a rotary wing aircraft can be reduced to zero. On this basis, a design for a system of equipment to maintain automatically a net charge about the zero level was evolved, and extensive operational tests conducted.

The corona discharge equipment constitutes a system designed to maintain a specific net charge on a helicopter. An electrostatic field sensor is used to measure the incident field due to the electrostatic charge on the aircraft. An "error" signal, proportional to the difference between the charge existing on the aircraft and a chosen reference level is used to control operational sequence and polarity of a high voltage power supply connected to a corona discharge point. The removal of charge from the aircraft, in the form of a discharge current, is done in such a manner as to minimize the error signal, thus a closed loop system is formed.

Upon completion of the prototype system equipment, initial local area flight tests in an H-21 aircraft were conducted. With this aircraft, it was not possible to obtain a sustained charge of sufficient magnitude to exceed system threshold levels, and normal operation in a hover attitude was not observed. The aircraft was flown in the vicinity of chemical plant exhaust smoke and over the Niagara Falls to experience a sufficiently large charge build-up. The disadvantage of limited operation in a variable field was apparent, but results were adequate to verify proper operation and to bring to light certain system deficiencies which were corrected.

Final flight tests were conducted at Edwards Air Force Base, California. System equipment was installed in an H-37 aircraft and numerous oscillograph recordings made of system action for various

aircraft altitudes. With the aircraft in a 25 foot hover altitude, the aircraft charge was reduced to the system threshold level in approximately $4\frac{1}{2}$ seconds. With the aircraft charge maintained between ±500 volts per meter (v/m), the resulting charge voltage to ground was measured to be on the order of + 300 volts. On the basis of reduced data, it is felt program aims were very well met.

II CONCLUSIONS

The flight test data presented and discussed in Section VII demonstrate that the automatic discharge system operated as was intended in its design. Rapid reduction of the aircraft charge potential to a zero level, and maintenance about this point within specified limits, was possible for aircraft attitudes of hover and forward flight at various altitudes.

Difficulties were encountered in the form of improper triggering action of control circuitry because of excessive vibration effects of the sensor unit. Remedial measures, in the form of more adequate isolation of the electrometer tube, corrected this malfunction. Modification to a second sensor unit, utilizing a solid state device in place of the electrometer tube, provided more operational information on a comparative basis especially with regard to system vibrations.

In a hover attitude at altitudes of 25 feet and 50 feet, time required to reduce the aircraft charge to the system threshold level was on the order of 4 to 6 seconds. A first cycle negative overshoot was experienced consistently with the field maintained between zero and +500 v/m thereafter.

With the aircraft in forward flight (60 knots I.A.S.) at altitudes of 500 feet and 1000 feet, both positive and negative overshoots occurred for almost each cycle of operation, because of the decreased aircraft capacity experienced at this altitude. Bipolar capabilities of system performance, and ability to maintain a field reasonably close to design limits, were displayed. Reduced data for this mode of operation are highly indicative of effects of system parameter variations and can be applied to problems associated with other aircraft.

It is concluded that the basic problem of effects of static charge build-up on a hovering H-37 aircraft have been successfully solved with the charge reduced to zero and maintained between predetermined limits. The level of reduced net charge is sufficiently low to enable ground handling personnel to discharge their duties during a cargo handling procedure without danger of a resulting shock.

III RECOMMENDATIONS

The flight test data of the automatic discharge system demonstrate that the method used satisfies the contract specifications. A discharge system has been provided which is adequate for maintaining an H-37 aircraft near zero charge, in a fair weather environment, to eliminate the hazard associated with cargo handling operations.

Improvements in overall system performance and configuration, from a standpoint of decreased size, simplification and added reliability could very well result from a redesign of this prototype equipment. Design goals could very well be; reduction in size to one cubic foot excluding probe and sensor and a total weight figure approaching 25 pounds.

Results of basic work during this program period have shown that added work is required to adapt the present system to other aircraft such as the HU-1, H-34, H-21 and YHC-1B. Information regarding helicopter capacity, charging rates and sensed fields versus charge voltage are required to enable a redesign of the basic system to be compatible with these various parameter sets. Provisions could be made whereby a basic system could be readily adapted by resetting gains and threshold levels along with proper choice of a plug-in type high voltage unit.

All flight testing was done in a fair weather ambient with complete lack of precipitation or snow. An appreciable design margin was incorporated into the present system, which should allow for extreme deviations from charge magnitudes experienced during field test activities; however, it is recommended that more effort be expended in determining charge build-up effects under more adverse conditions.

IV INTRODUCTION

A. STATEMENT OF PROBLEM

The phenomenon of static charging of aircraft has been a source of interest to aircraft operations since the early days of aviation. Originally the major concern was the radio interference caused by the erratic and violent static discharge from fixed wing aircraft in flight. Later, this phenomenon came to represent a potential fire hazard during airborne refueling operations. With the advent of rotary wing aircraft and the external carriage of cargo, the existence of high voltage static charge build-up led to new problems of injuries to field personnel engaged in the ground handling of loads attached to a hovering helicopter. Additional possible hazards are reactions of certain cargo, such as fuel, warheads, ammunition, etc., to the flow of electrical energy when the discharge action takes place on ground contact.

B. CONTRACT SPECIFICATIONS

Cornell Aeronautical Laboratory, Inc., was authorized by the Transportation Research Command to design, develop, and fabricate an experimental model of an automatic corona discharge system applicable to several helicopter types. Project effort was originally divided into three separate phases in accordance with the Statement of Work of the Contract. During the course of the program, modifications were made to the contract for additional work categorized as Phases IV and V.

Phase I consisted of the preliminary design of an automatic corona discharge system based upon the concept developed under Contract DA 44-177-TC-544 (Ref. 2). This system has been further refined to meet the following technical specifications:

1. The system shall be fully automatic such that no personnel other than the normal flight crew shall be required for operation.

2. The size of the system shall be the minimum possible, not to exceed 2 cubic feet, excepting the corona probe.

3. The weight of the system shall be the minimum possible, not to exceed 50 pounds.

4. The equipment shall operate on the power available from the existing electrical system of the aircraft.

5. The equipment shall be adaptable for use on the following Army helicopters: H-21, H-34, H-37, HU-1 and YHC-1B.

6. The system shall operate properly in all environmental and weather conditions compatible with helicopter operation.

7. The system shall not be of such a nature that issuance of extra equipment to ground personnel is required.

8. The system shall not adversely affect radio and/or navigation equipment of aircraft on which system is installed.

Phase II consisted of the procurement of parts and the construction of experimental components for the system. One complete set of units was assembled and laboratory tested to verify design performance. Local area flight tests were made with the system installed in an H-21 aircraft.

In Phase III the equipment was installed in a government-furnished H-37 aircraft, and flight tested in a hot, dry environment. Data were obtained on the system characteristics under these environmental conditions.

Phase IV consisted of fabrication of a duplicate set of equipment components to be installed in an H-37 aircraft at Ft. Eustis, Va. These items were completed in time to be flight tested during the Phase III portion of the program.

In Phase V, the original sensor was modified to be used as a basic field measuring device for extended ranges to 100 kv per meter.

V TECHNICAL DISCUSSION

A. AIRCRAFT CHARGING PHENOMENON

Although the phenomenon of static charge has been associated with aircraft since the inception of such vehicles, and while the process leading to charge generation is well understood, there is no simple method available for calculating the potential of an aircraft in flight relative to its surroundings. There are several reasons of this; first, there is a wide variety of ambient conditions possible over which there is no control and the instrumentation to measure these conditions is difficult if not at times impossible. Secondly, the total charge is a combination of several mechanisms operating at the same time.

A helicopter may accumulate an electrostatic charge of several thousand volts during normal flight conditions from at least three causes. First, the rotor blades are large wing areas rotating at high peripheral speeds and the resulting contact between these wing surfaces and the air containing particles of dust, sand, smog, etc., is a major cause for this static build-up. Secondly, the charged particles emitted by the engine exhaust result in a charge build-up opposite in polarity to that of the emitted ions. The third process is induction. An aircraft flying through large variations in the earth's electric field, such as in the vicinity of highly charged clouds, will assume a polarization opposite in sense to the field to which it is exposed.

A major advantage of rotary wing aircraft is the ability to hover over a fixed point during cargo loading or unloading operations. In this attitude, static charge voltage presents a serious problem to ground handling personnel. When contact is made with the cargo or cargo hook of a hovering helicopter, the human body acts as a discharge path to ground and the resulting shock received can be hazardous. The secondary effects can also be quite serious in that one may inadvertently make a motion which might place himself, other ground personnel, or the helicopter in a hazardous position. Certain type cargo, such as detonators, ammunition, etc., may also respond to this discharge current in an unfavorable manner.

B. PREVIOUS WORK

Under Contract DA 44-177-TC-544, Cornell Aeronautical Laboratory conducted studies and feasibility tests to determine the relative merits of various means to eliminate the static charge build-up on a hovering helicopter (Ref. 2).

Use of insulating type load cables to isolate the load from the helicopter proper was found to be unsatisfactory from both a mechanical and electrical point of view. Characteristics of nylon, which was found to be the best of the materials investigated, are variable with changes in relative humidity and temperature. The important variables are tensile strength and ohmic resistance. With an insulated cable, a charge proportional to the cargo surface area can exist, thereby recreating the same problem.

Wick type static dischargers mounted on the main rotor tips of an H-37 were found to be effective in reducing the initial charge on the helicopter by approximately 50%. However, since the static charge process itself must supply the charge requirements necessary to cause these devices to function, charge can be reduced only to some threshold value which for the H-37 helicopter was significant.

Characteristics of reciprocating engine exhaust effects and contribution to the overall charge on the aircraft were investigated. The effect of the ions present in the exhaust gases was to charge the aircraft negatively to approximately 10% in magnitude compared to that experienced in a hover attitude. The polarity, being opposite in sign to that normally resulting from rotor action, tends to reduce the peak charge build-up. The rate of ion generation is a function of the throttle setting and most likely of the fuel composition, engine condition, and cooling air purity. A discharge technique based on such variables does not appear to be practical at this time.

A radioactive probe, which consisted of 2.5 millicuries of Radium 226 as an ion generator, was evaluated as a discharge device. The source size used was not large enough to reduce adequately the charge to a safe limit but did provide operating characteristics and discharge information necessary for calculation of a sufficient source size. This system offers advantages in that it is passive in operation and is very small in size; however, the contamination dangers associated with a source size large enough to reduce the charge sufficiently are restrictive, relative to storage, handling in normal operation or in the event of an aircraft accident. On this basis, the system was considered unsatisfactory.

A high voltage corona discharge system was proven to be the most practical and effective means to neutralize the static charge build-up. Flight tests at various altitudes and aircraft conditions pertinent to the cargo handling problem were conducted with various probe and sensor locations to determine an optimum configuration. Other tests were conducted to determine charging rates and time constants needed for design of an automatic system.

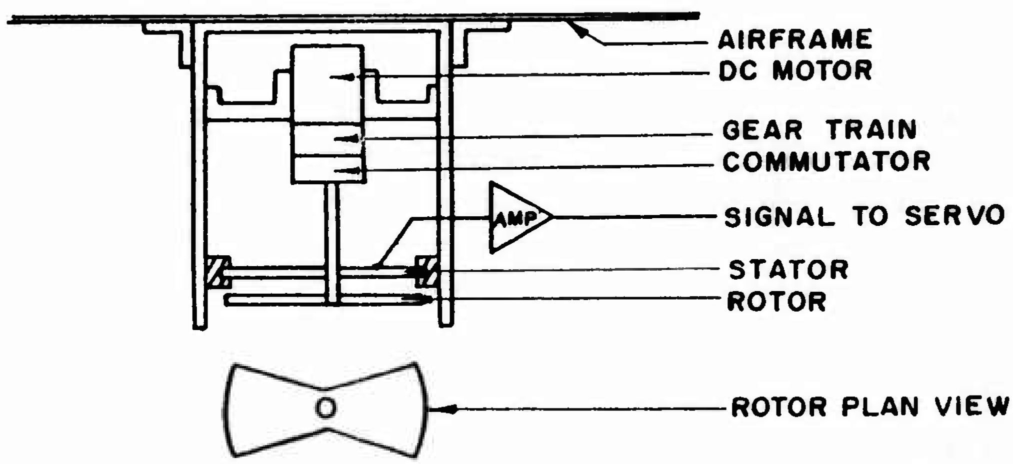
C. BASIC SYSTEM

The corona discharge equipment constitutes a system designed to maintain automatically a specific net charge on a helicopter. An electrostatic field sensor is used to measure the field due to electrostatic charge on the aircraft. An "error" signal proportional to the difference between the charge existing on the aircraft and a chosen reference level is used to control operational sequence and polarity of a high voltage power supply connected to a corona discharge point.

The removal of charge from the aircraft, in the form of a discharge current, is done in such a way as to minimize the "error" signal; thus a closed loop control system is formed. A block diagram of the system is shown in Figure 1. Figure 2 is a sketch of equipment installation in an H-37 aircraft. Elements of this system are discussed in detail in the following sections.

D. SENSOR - GENERATING VOLTMETER

A generating voltmeter is used as the sensor to measure the magnitude and to sense the polarity of the surrounding electrostatic field. The basic elements of the generating voltmeter are depicted in the following sketch.



The stator plate is isolated from case ground by a teflon dielectric support and is alternately exposed to and shielded from the electrostatic field as a function of the rotating vane position. A planetary gear reduction assembly is used to obtain an output shaft speed of 20 revolutions per second from the basic 6000 revolutions per minute drive motor. Two signal samples are sensed per revolution for an

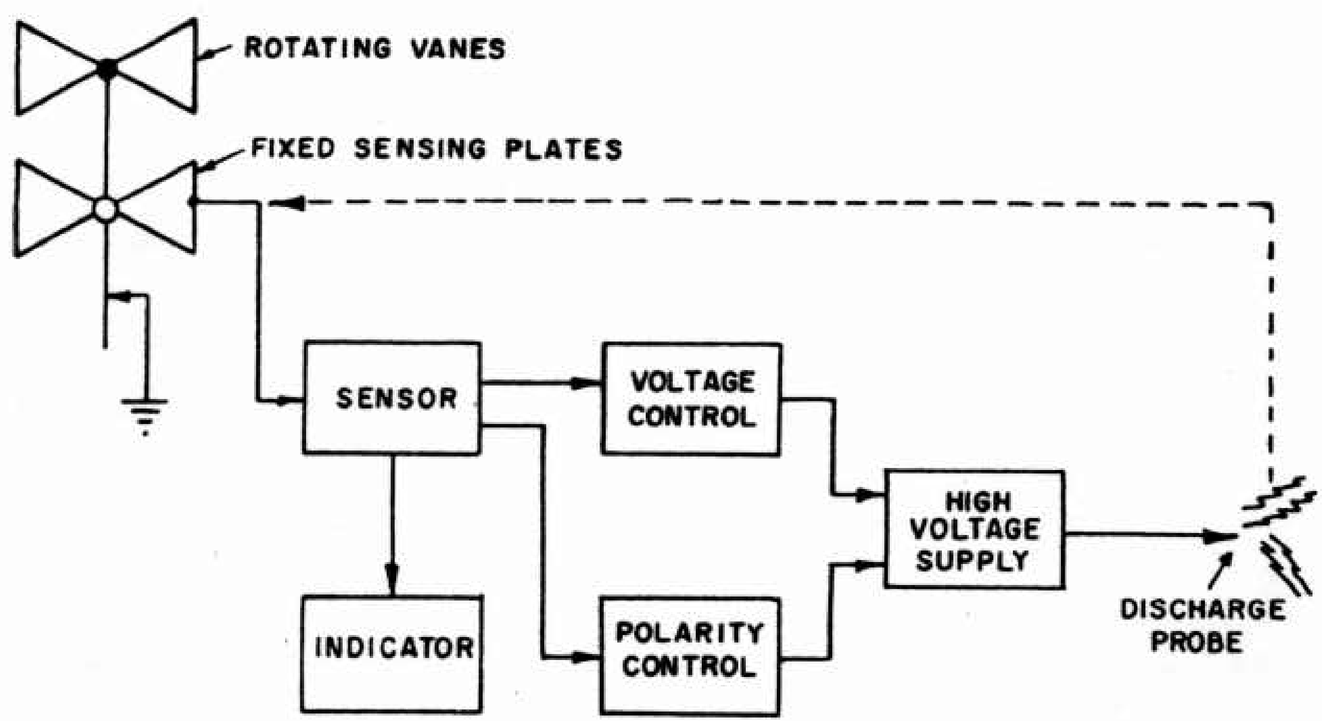


Figure I SYSTEM BLOCK DIAGRAM

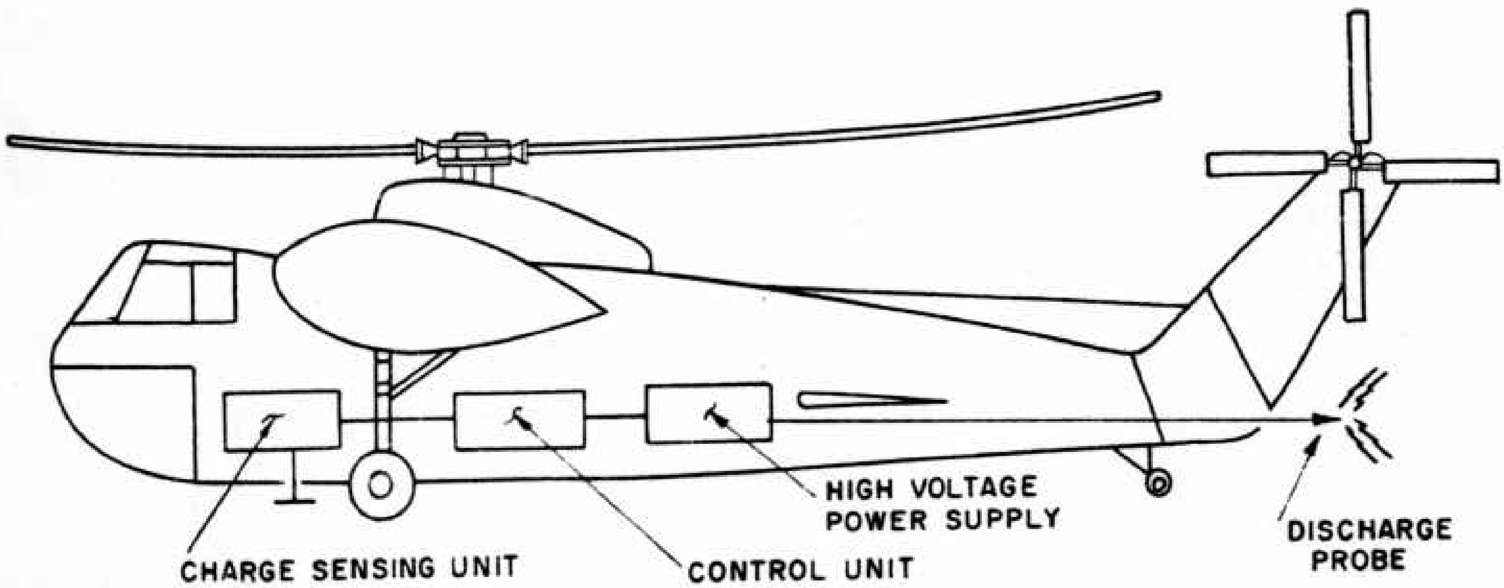


Figure 2 SYSTEM INSTALLATION, H-37 AIRCRAFT

output frequency of 40 cycles per second. The charge built up on the capacitive stator plate is the signal input to the amplifier and is sinusoidal in character; however, since the amplifier output is grounded by means of commutator action during the time interval that the vane is passing through the shielded position, the sinusoid is converted to a half wave function. The peak value and polarity of this signal is then a quantitative measure of the sensed field after calibration in a known, controlled field has been made.

The generating voltmeter used for field measurements during the feasibility program (Ref. 2) was designed primarily for laboratory usage and a redesign of the unit was deemed necessary in the interests of increased reliability, and performance compatible with the automatic control system requirements in the aircraft environment.

A schematic diagram of the sensor electronics appears in Figure 3. The signal amplifier consists of four transistorized stages with an overall gain of approximately 2000 at 40 cps operating into a 2000 ohm load. Input impedance on the order of 40×10^6 ohms is achieved through use of a "field effect" transistor (C 614) operating in a triode mode. Gain of this stage is approximately 4 with inherent noise down to an extremely low level. The following two stages are a standard common emitter amplifier driving an emitter follower which provides adequate isolation along with relatively low output impedance. The sensor amplifier output is approximately 1.5 volts peak for a field of 500 volts per meter. A diode limiter between base and emitter of the last stage limits the output signal to a peak value of ± 5 volts.

The commutator characteristics also appear in this figure. The brush on segment 1 provides an "on" reference signal to the controller during specific portions of the rotor cycle. Segment 2 grounds the shaft and rotor and maintains the rotor at case potential. Segment 3 provides a reset pulse to the sequential servo through a separate emitter follower transistor stage.

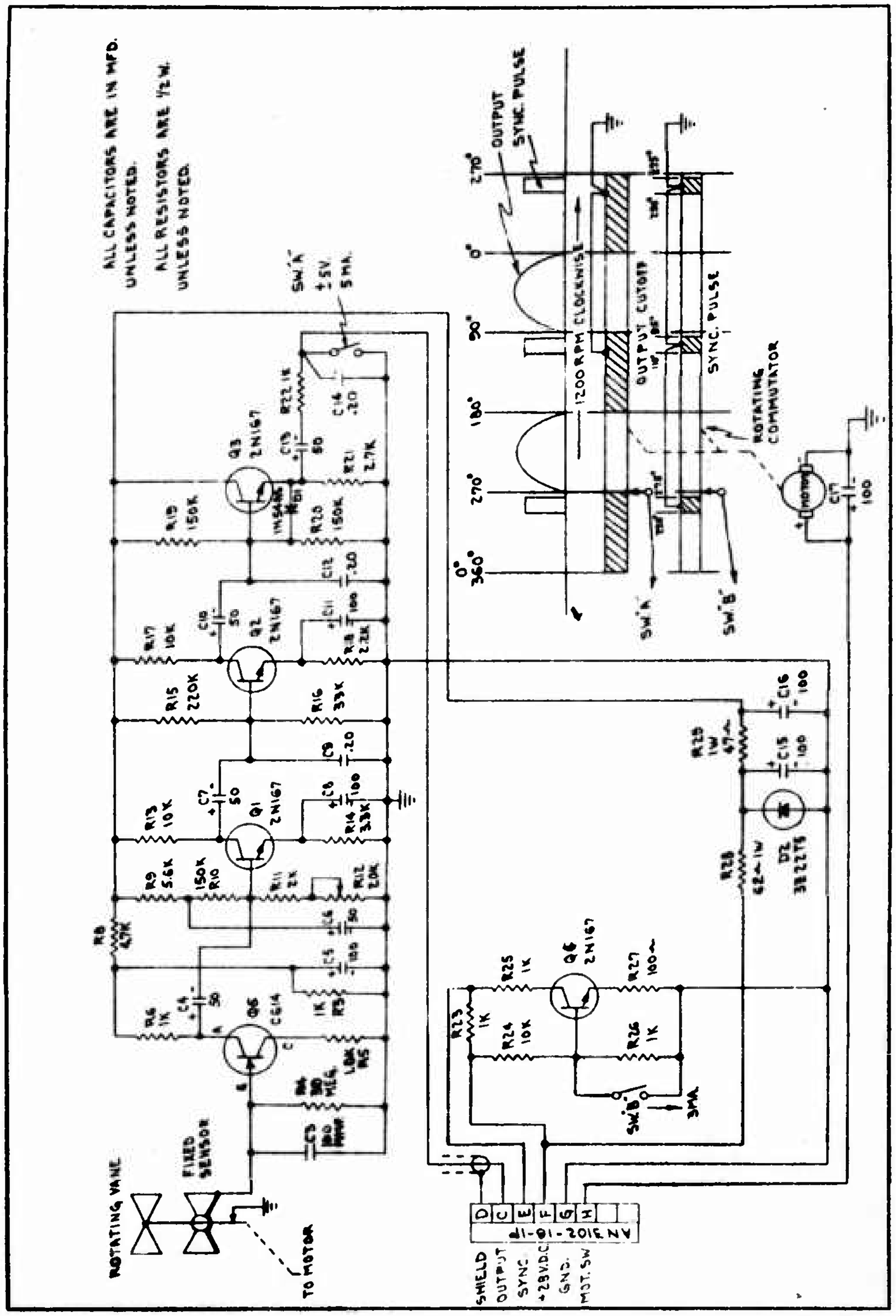


Figure 3 SENSOR SCHEMATIC (STATIC CHARGE)

Figure 4 is a photograph of the sensor unit showing a breakdown of component assemblies along with an assembled unit with case removed.

E. CONTROL SYSTEM

In order to design an effective control unit for the automatic discharge system, it was necessary to determine three prime factors: first, the range of values to be measured, and form of presentation to the control system; second, the requirements demanded by the controlled elements; and third, the limits of arbitrary net charge to be maintained for personnel safety.

The fact that charge buildup on a helicopter can assume either polarity, with respect to earth, dictates bipolar requirements for the control system. Sensor output, used to actuate control circuitry, is a series of half sine waves referenced to sensor ground with peak amplitude proportional to the level of, and same polarity as the incident field.

The controlled element is a high voltage, fixed level power supply capable of bipolar operation with respect to sensor ground. In the interests of size, weight, and relative economic considerations, a decision was made to utilize a single unit with a polarity switching device in lieu of a dual supply. No foreseeable degradation of system performance would be imposed due to this configuration. The power supply has the capability of being controlled by sequential type operation.

Laboratory experiments have shown that a 200 micro-microfarad capacitor charged to a potential of 1500 volts gives no electrical sensation when discharged through the human body between relatively moist hands. A slight sensation is felt under the same conditions with a 2000 micro-microfarad capacitor charged to 900 volts. Assuming a maximum capacity of 1000 micro-microfarads for an H-37 in a hover attitude at an altitude of 25 feet, a decision was made to limit threshold levels of arbitrary net charge voltage to ± 500 v/m for an

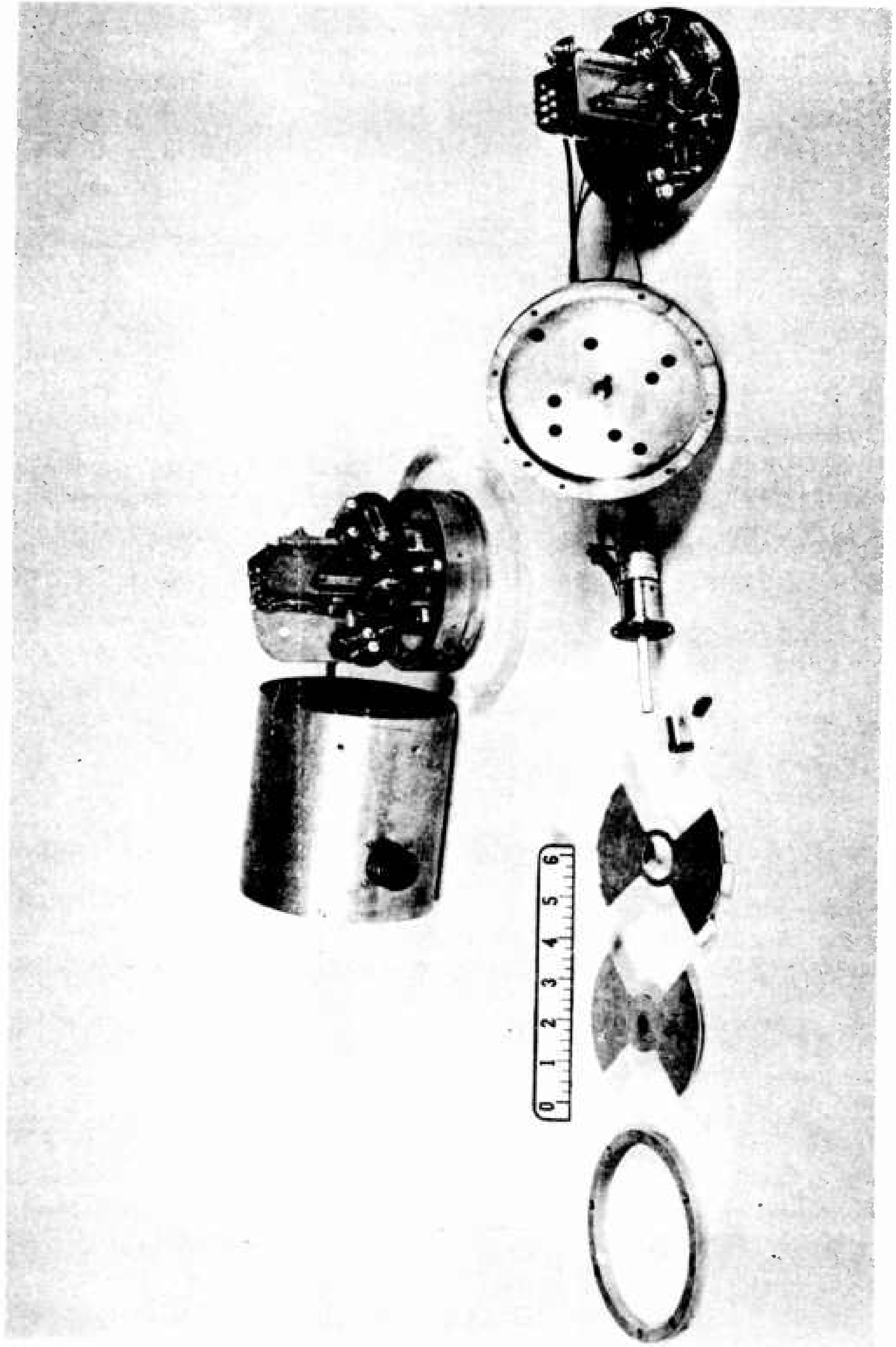


Figure 4 COMPONENT ASSEMBLIES, SENSOR UNIT

indicated zero condition. Energy levels for other type helicopters would be proportional to capacity of these aircraft relative to ground in a hover attitude.

Three basic types of control systems were considered: Analog, Proportional Pulse Width, and Constant Error Sequential. Solid state devices, such as transistors and diodes, were to be used in lieu of vacuum tubes.

An Analog System, in particular a solid state DC control system, is subject to large errors due to drift of operating point with varying environmental conditions, internal characteristics, etc. In order to compensate for these shortcomings, complex circuitry would be required. Another problem requiring added circuitry would be provision for linear operation over the range desired. Switching sequence, from one polarity to another, becomes a problem when pure proportional control is considered.

A Proportional Pulse Width Control System, using a logical switching design approach, provides saturated output for a time that is proportional to the amplitude of the actuating error signal. Threshold error levels are preset and power supply polarity is predetermined on the basis of polarity of the sensed field. This approach alleviates a good percentage of the drift problems present in the analog system, but certain portions of the circuitry would still have inherent drift problems.

A Constant Error Sequential Control System operating on a pulse basis and utilizing logical switching design also takes advantage of a saturated output. When the power supply polarity is correct and actuating error is sufficiently high, control of high voltage is effected at a constant pulse rate. Drift problems are essentially negated since all individual circuits are in either of two states: saturated or cut off.

A comparative analysis was made of the three types of control systems with the necessary assumptions for the various aircraft under consideration. On the basis of time response, maximum possible error, reliable operation and ease of maintenance, it was decided the Constant Error Sequential Control System would offer the best approach. The following operational philosophy refers to this system.

1. General

The input signal to the control system is a half sine wave whose polarity with respect to the helicopter is either positive or negative, based on the incident field of the sensor. This input is noted by (S_o) on the block diagram of the control system, Figure 5. The peak amplitude of this signal is proportional to the strength of the induced field. When this signal, which occurs forty times per second, reaches, or is greater than, a predetermined threshold, a trigger pulse is initiated (S or S'), depending on input polarity. A comparison is then made between this trigger and the polarity bistable multivibrator as to the polarity of the power supply with respect to the airframe (P and $\overline{P}$). If the polarity is compatible with the trigger, the amplitude bistable is switched to the correct level for the power supply "ON" condition. A noncompatible polarity with the trigger would result in the trigger switching the polarity bistable to the correct state and at the same time initiating a monostable delay hold (M) which inhibits trigger pulses from activating the power supply for a finite time. This delay insures that the power supply remains off during polarity reversal sequence. The amplitude bistable is switched off by a synchronizing pulse (R) immediately prior to the time an amplitude trigger pulse is expected. If the trigger appears, the duration of time that the amplitude bistable would be in the off condition would be less than the response time of the amplitude portion of the power supply and as such would be relatively insignificant. Figure 6 is a photograph showing both top and bottom view of this assembly.

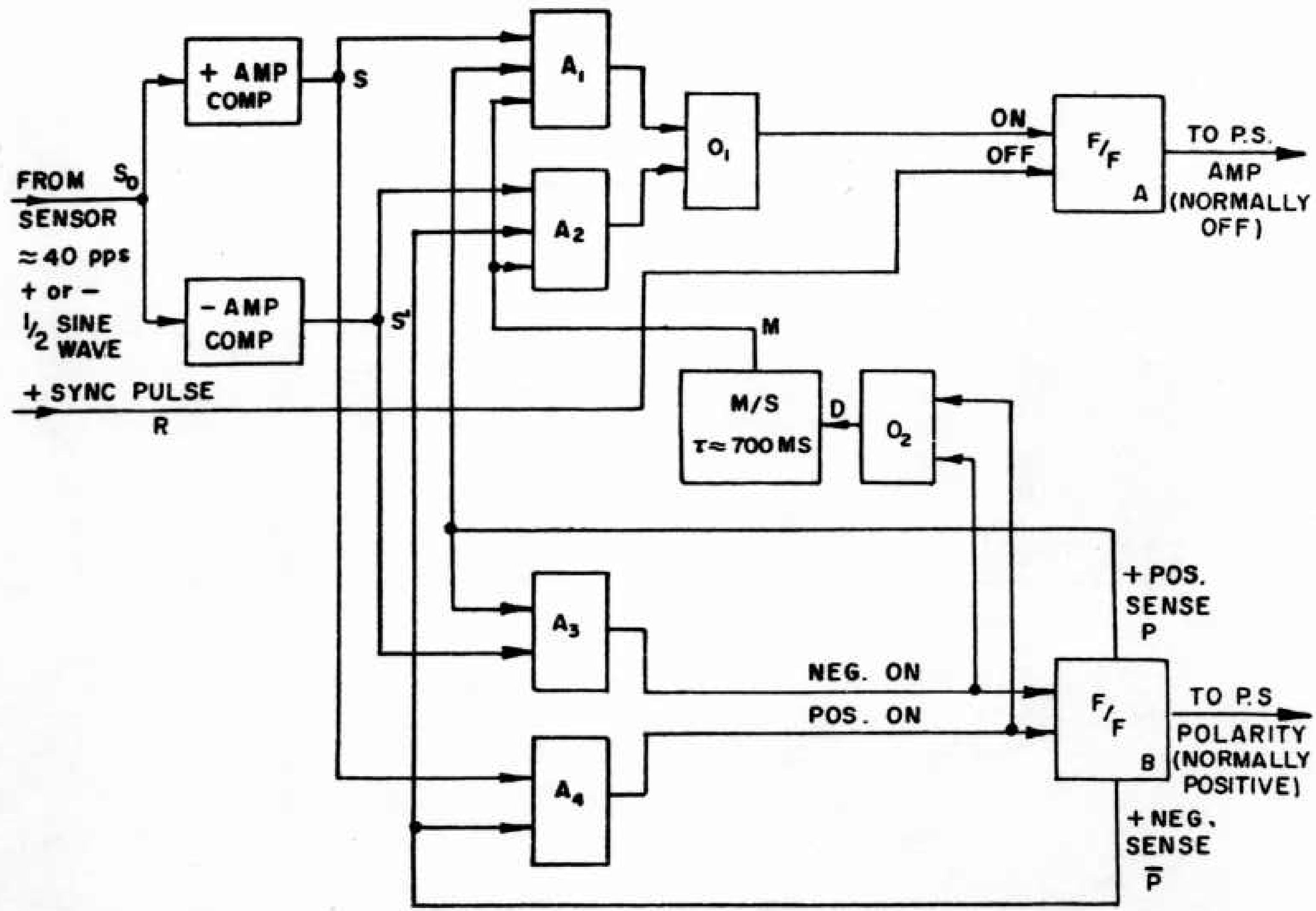


Figure 5 BLOCK DIAGRAM, STATIC CHARGE CONTROL SYSTEM

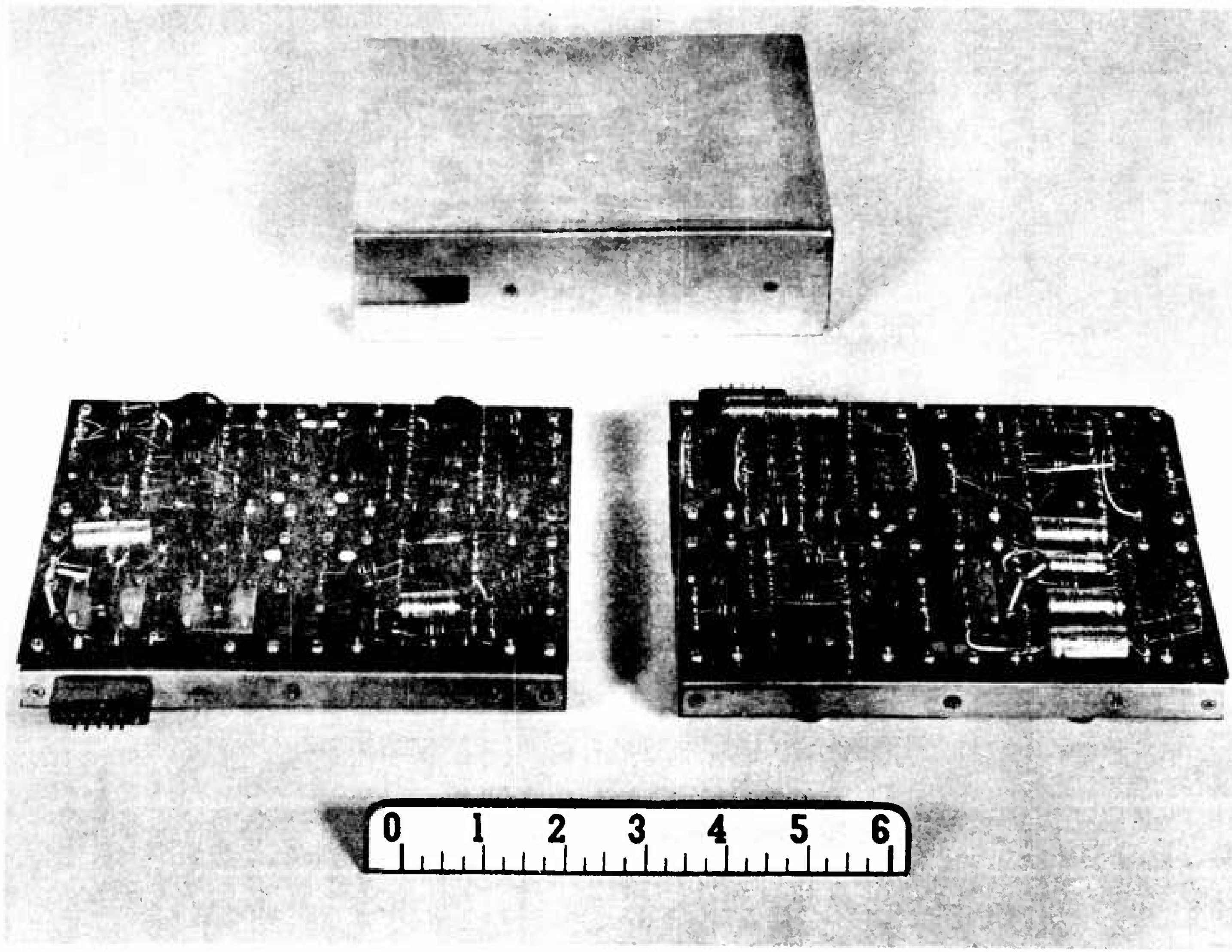


Figure 6 CONTROL UNIT ASSEMBLY

2. Amplitude Comparators

There are two Schmitt trigger amplitude comparators in the control system. One accepts the input directly and delivers a trigger output (S) when the input amplitude is above the threshold in a positive direction. The other accepts the input after it has been inverted and the combination inverter and comparator emits a trigger (S') when the input is negative and the amplitude is greater than the threshold level. The threshold of each can be separately adjusted above ground. This level or dead zone below which no action occurs is an aid, in that for some system overshoot, polarity of the power supply will not be reversed. The output of the amplitude comparators is transmitted to the logic circuitry.

3. Amplitude and Polarity Bistable Multivibrators

Both the amplitude and the polarity bistable multivibrators are conventional solid state saturating type multivibrators with separate "ON" and "OFF" positive triggers required.

The amplitude bistable accepts "ON" pulses from the logic circuitry and "OFF" pulses from a synchronizing signal. The operation of this bistable is such that a synchronizing pulse arrives just prior to the time that an amplitude pulse would arrive if the amplitude pulse were present. This assures that the amplitude bistable is "OFF" if no amplitude pulse arrives. If consistent amplitude pulses arrive, the "OFF" time of the bistable is sufficiently short so as not to interfere with the power supply. The output of this bistable would go to an emitter follower and then to the pass transistor in the power supply. Output is normally off.

The polarity bistable accepts both "ON" and "OFF" pulses from the logic circuitry and both outputs from the polarity bistable are returned to the logic circuitry as polarity indicators. Output is also given to an emitter follower which controls the polarity relay. The output is normally off, indicative of positive polarity.

4. Delay Monostable Multivibrator

The delay monostable multivibrator is used as a hold device such that the power supply is maintained in an off position for the duration of time required for the polarity relay to switch. The input to the delay monostable is from the logic circuitry and originates at the inputs to the polarity bistable. Output of the monostable is to the "and" circuits following the amplitude comparators and constitutes an inhibit function.

5. Logic Circuitry

The logic circuitry is formed by the use of diode logic. The logic dictates under what conditions the power supply may go on and what polarity is required. The logical equations used are as follows:

Restriction on Amplitude (inherent)

$$(SPM + S'\bar{P}M)\ R = 0$$

Amplitude ON

$$SPM + S'\bar{P}M = 1 \qquad (1)$$

Amplitude OFF

$$R = 1 \qquad (2)$$

Restriction on Polarity (inherent)

$$(S\bar{P})\ (S') = 0$$

Polarity Positive

$$S\bar{P} = 1 \qquad (3)$$

Polarity Negative

$$SP = 1$$

Time Delay

(4) $$SP + S'\overline{P} = \overline{M}$$

Where S = Positive Amplitude Comparator Output

S' = Negative Amplitude Comparator Output

P = Positive Polarity Output

$\overline{P}$ = Negative Polarity Output

M = Positive Monostable Output

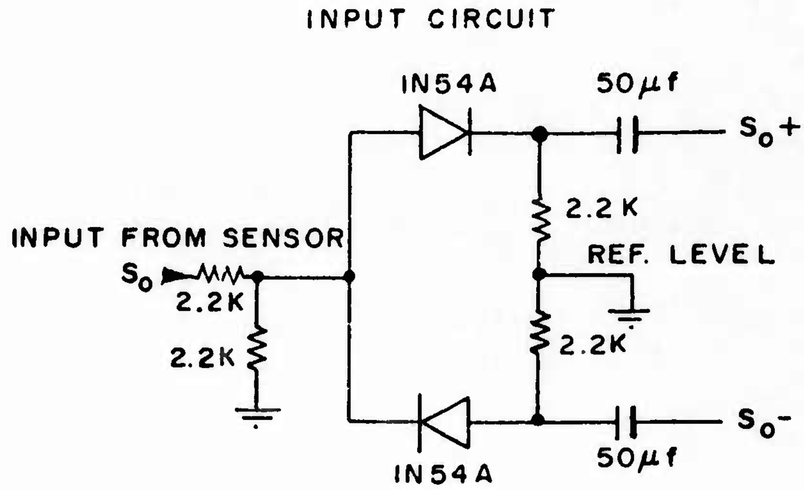
$\overline{M}$ = Negative Monostable Output Lasting for approximately 100 ms and Reverting Back to M

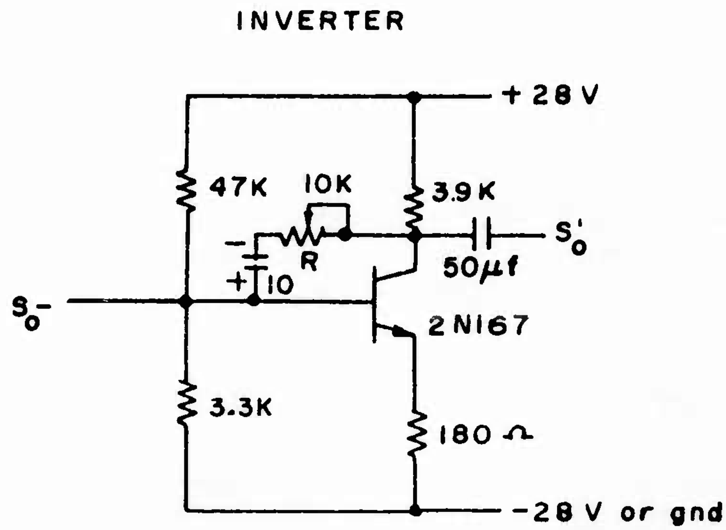
Various circuit schematics of logic circuitry appear in Figures 7, 8 and 9.

F. HIGH VOLTAGE POWER UNIT

A survey of the AC and DC primary power available for the static discharge system and instrumentation was made for the H-21, H-34, HU-1, H-37 and YHC-1B helicopters. The H-34, H-37 and YHC-1B aircraft have more than sufficient AC and DC power to meet system requirements; however, the 115 volt 400 cycle margin on the H-21 and HU-1 aircraft is approximately 50 volt amperes which could easily be depleted by a change in normal complement of flight and/or navigational instruments. As a result of these findings, it was decided to consider the nominal 27 volt ± 1/2 volt DC as the sole power source available.

On the basis of field tests with the H-37 aircraft and taking into consideration a nominal overdesign factor, voltage requirement for the corona discharge system was limited to 25 kilovolts (KV) maximum. Discharge current requirements for this system are on the order of 20 microamperes, indicating the supply used, which is capable of one milliampere total output, is sufficient.

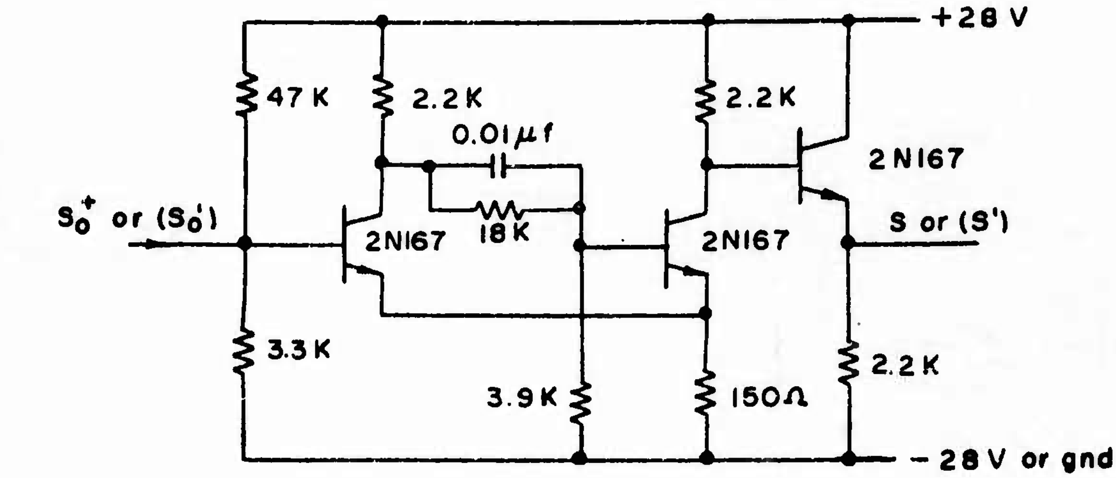




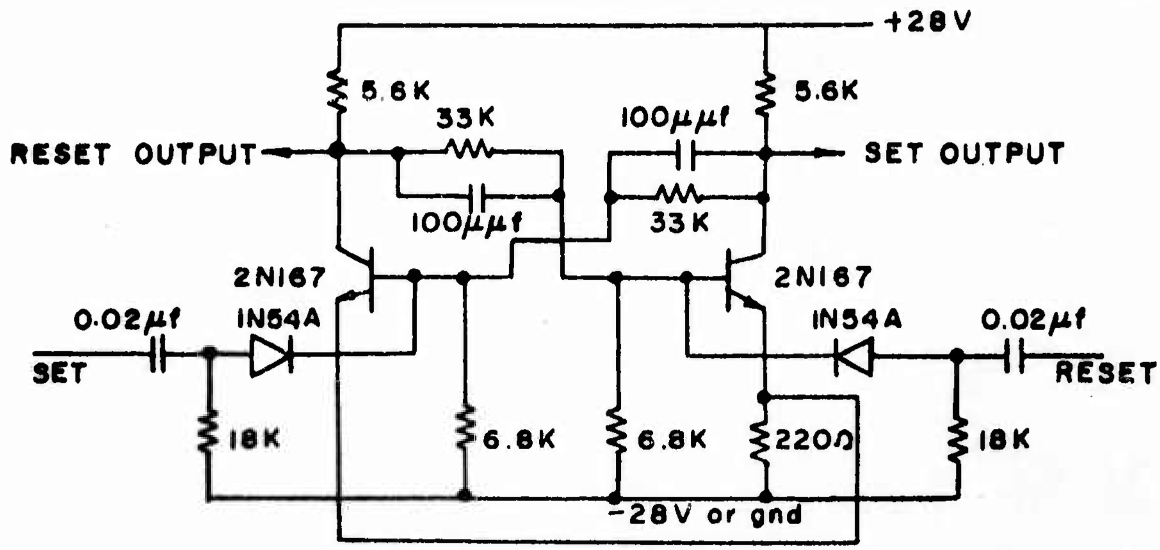
R is a 10K trimpot adjusted such that S_o' in normal loaded condition is of opposite polarity but equal amplitude to S_o- (≈6.8K)

Figure 7 CIRCUIT DIAGRAMS, INPUT CIRCUIT AND INVERTER

SCHMITT TRIGGER



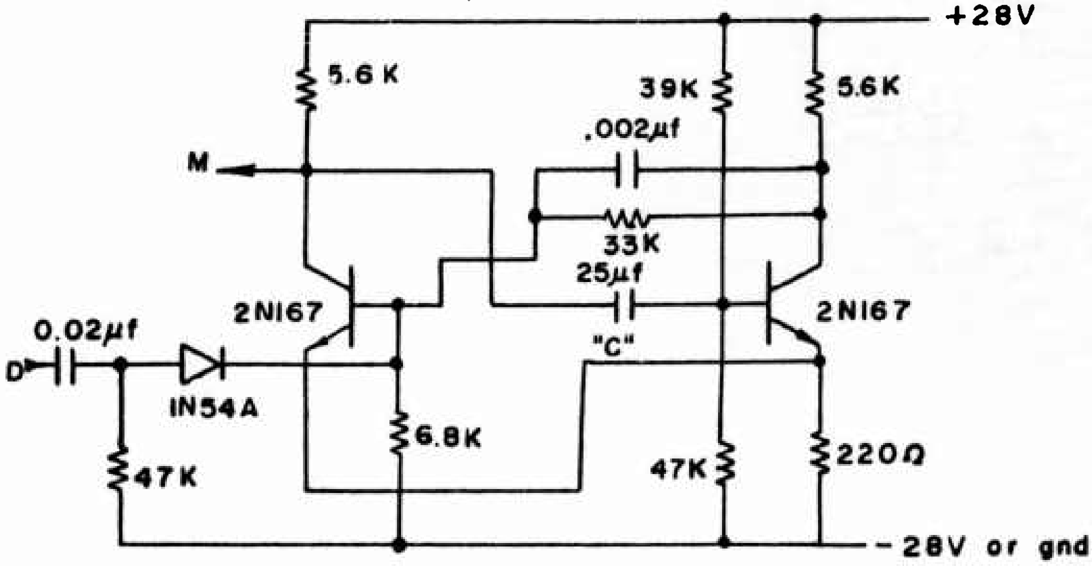
BISTABLE MULTIVIBRATOR



AMPLITUDE BISTABLE SET → ON , SET OUTPUT → HIGH
RESET → OFF , RESET OUTPUT → HIGH
POLARITY BISTABLE SET → NEG ON, SET OUTPUT → HIGH ($\bar{P}$)
RESET → POS ON, RESET OUTPUT → HIGH (P)

Figure 8 CIRCUIT DIAGRAMS, SCHMITT TRIGGER AND BISTABLE MULTIVIBRATOR

MONOSTABLE MULTIVIBRATOR



Capacitor "C" equal to 5µf offers a delay of about 150 ms. at M.

LOGIC CIRCUITRY

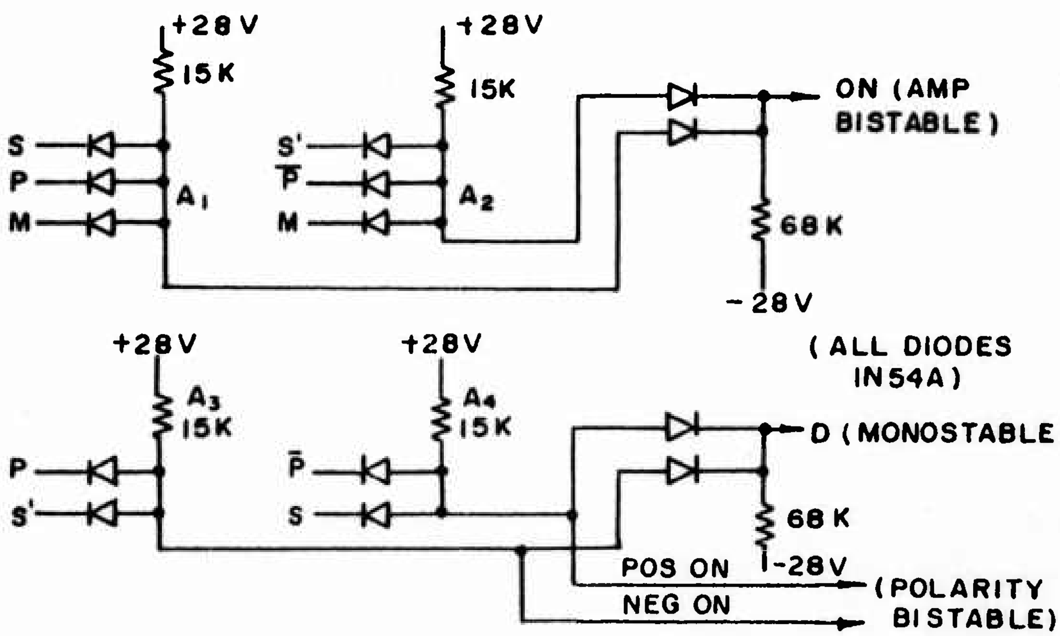


Figure 9 CIRCUIT DIAGRAMS, MONOSTABLE MULTIVIBRATOR AND LOGIC CIRCUITRY

The high voltage power supply block diagram is shown in Figure 10. The aircraft 28 volt DC is filtered and then fed into a transformer coupled transistor multivibrator which develops an essentially square wave output at 1 kilocycle (KC). Two secondary windings of the input transformer terminate in the base and emitter of two medium power transistors connected to the primary winding of the high voltage transformer as a common collector amplifier. The forward bias for these transistors is furnished through full wave rectification of additional secondary windings of the input transformers.

A magnetic amplifier regulator is employed, with the AC windings fed by the input transformer secondary and the DC control winding signal originating in a voltage comparator with a zener reference. The rectified AC winding output is summed with the rectified output of additional windings and is regulated within 1% of a nominal 23 volt DC average value.

A pass transistor remotely controlled by the sequential servo on-off bistable is in series with this pulsating DC voltage and in the "ON" position, this voltage is applied to the center tap of the high voltage transformer primary providing a pulse modulation to the common collector amplifier.

The oil immersed high voltage section is comprised of a voltage quadrupler, bleeder resistors and a double pole double throw polarity reversing relay remotely operated by the control servo. The output voltage is 25 KV at 1 milliampere with an input power requirement of 2 amperes at 28 volts DC.

The unit size is approximately 8 inches high x 9 inches wide x 12 inches long with flange type mounting provided. The front end components are spread out in open board construction to permit easy access for checking and replacement. With the solid state components used, the power supply would be amenable to standard aircraft electronic packaging.

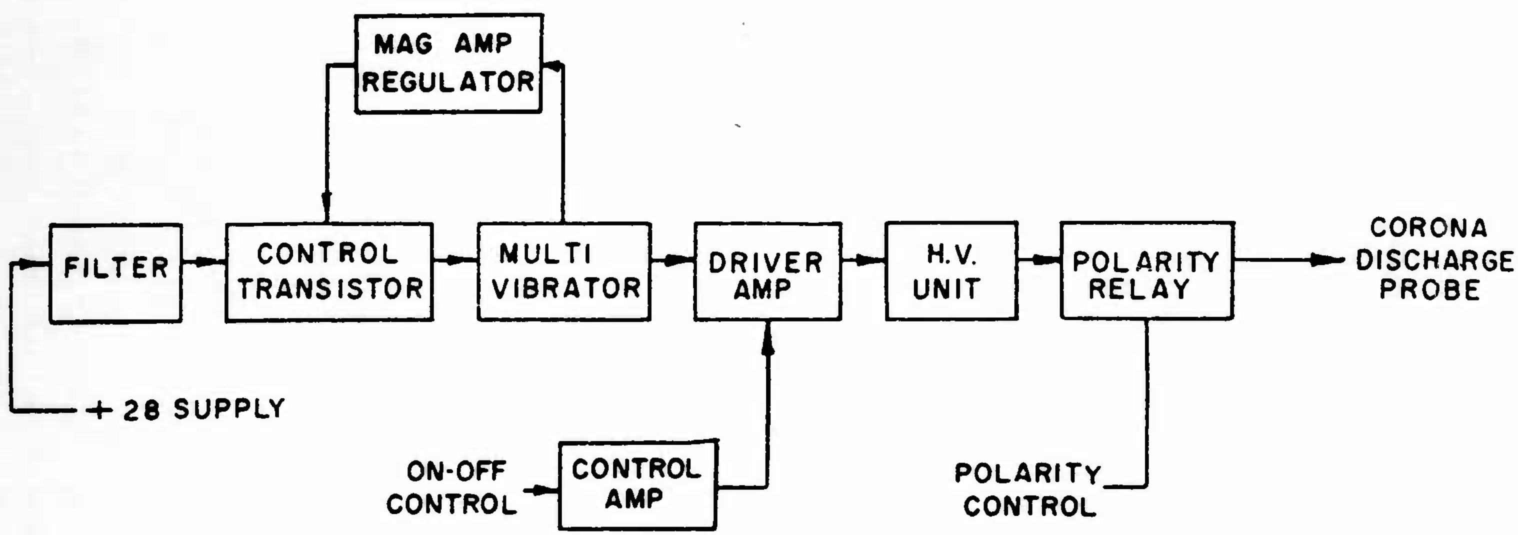


Figure 10 BLOCK DIAGRAM - HIGH VOLTAGE POWER SUPPLY

G. DISCHARGE PROBE

A primary requirement of the corona discharge system is to insure that the current discharged from the corona point does not travel back to the aircraft but is blown away by the main rotor slipstream. Consideration of this problem indicated that locating the corona point at least 15 centimeters behind the aircraft guarantees that ions with a normal mobility at sea level of two centimeters per second, per volt, per centimeter cannot reach the aircraft when downwash velocities greater than 75 knots are present. This statement can be verified by considering the maximum voltage gradients which can exist near the probe and computing the component of ion velocity which opposes drift due to the slipstream. With 25 KV applied to the corona point, the average field along the insulation of the probe is 25,000/15 = 1667 volts/centimeter and the ionic mobility is 2.0 x 1667 or 3334 centimeters/second or 33.34 meters/second which is less than 75 knots. Sikorsky engineers have indicated main rotor downwash in the tail section area of an H-37 is in excess of 100 knots.

In addition to satisfying the requirements for minimum return of ions to the aircraft, the corona point must be so oriented that there will be no arc back to the aircraft skin itself. Careful design of the probe and mount assembly was made taking into consideration such problem areas as electrical breakdown from conductor to airframe, large surface resistance, and breakdown of air between point and airframe with maximum voltage applied. Figure 11 is a sketch of the probe design, with a photograph of three completed units, of different lengths appearing in Figure 12.

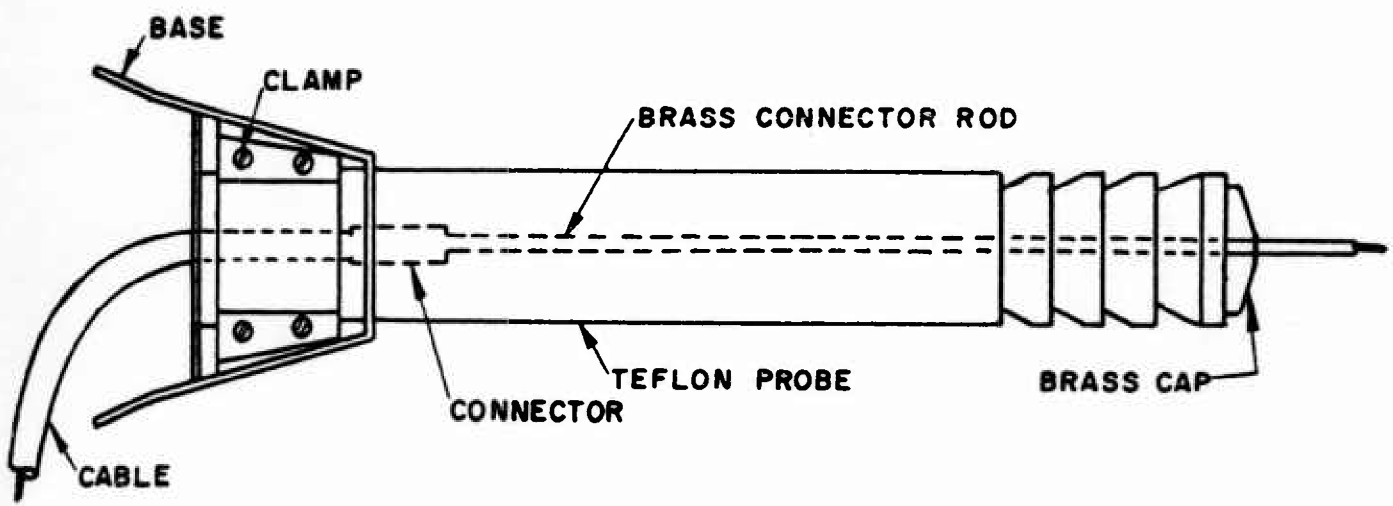


Figure 11 DISCHARGE PROBE

Figure 12 DISCHARGE PROBE ASSEMBLIES

VI SYSTEM TESTS

A. LOCAL AREA FLIGHT TESTS

In order to insure proper operation of the discharge system prior to commencing a field test program at a remote base, the following test sequences were made:

1) Bench type tests to determine operating characteristics under controlled conditions.

2) Airborne tests in an H-21 helicopter to determine characteristics under similar conditions to be experienced in the field.

During the normal development phase of the program, numerous bench tests were conducted to resolve operational problems. Most of the difficulties encountered were due to faulty components, and gain spread of the standard 2N167 transistor.

An H-21 aircraft was made available to the program, through the efforts of TRECOM, for initial local area flight tests. The aircraft, serial No. 562147, was supplied by the Aviation Section, Headquarters 2nd Artillery Group, Fort Niagara, New York.

Primary aims for this portion of the program were as follows:

1) To investigate equipment characteristics and operating ability with regard to aircraft attitudes and vibration characteristics.

2) To correct obvious equipment faults indicated by test results.

3) To obtain a level of confidence through actual typical operating conditions prior to extensive field tests at a remote base.

Local area flight tests were conducted during the period 2-28-61 to 3-22-61. A total of $8\frac{1}{2}$ hours of airborne time was logged and results indicated basic aims were accomplished.

Charge buildup in the H-21, on the order of +200 v/m, was found to be consistent regardless of variable ambient temperature and humidity for both the hover and forward flight altitude. In order to exceed the minimum threshold limit of approximately ±300 v/m, it was necessary to fly in the vicinity of charged areas such as chemical plant exhaust smoke and over Niagara Falls. Unfortunately, it was not feasible to hover in these areas to experience a sustained charge. As the aircraft approached the charged area, indicated field strength increased in the positive direction due to a change in reference sensing level from an assumed near zero to a highly negative value. An induced negative charge is experienced during the exposure period and as the aircraft moved into an area where zero reference level is experienced, indicated charge is negative for a period required for the normal charge mechanism to recharge the aircraft to nominal +200 v/m value. The main problem was to observe system action during the limited time interval (approximately 5 seconds) before the zero transition point was reached.

A quantitative measure of field strength greater than ±1000 v/m is not possible with system equipment, in that a saturated sensing level is reached at this point. This aspect, although compatible with system operation, was a distinct disadvantage with regard to knowledge of peak field strengths experienced. Preliminary flights over the Niagara Falls at an altitude of 1000 feet induced a charge much greater than the saturation level of the sensor and, although system functions were readily seen, the change in field effected with 25 KV on the probe resulted in a field which was still within the saturated region. An increase in altitude reduced the initial charge sufficiently so that effective field change to the threshold level was possible.

A reduction of CEC records for the first few flights indicated erratic system operation in that both positive and negative control signals were present regardless of input signal polarity. This condition was more apparent when a low level field was being sensed and it was found that both positive and negative control signals of sufficient amplitude to trigger control circuitry were present. Vibration characteristics of the aircraft were of sufficient magnitude to excite the electrometer stage of the sensor amplifier. This portion of the sensor unit was repackaged and successfully isolated to overcome this difficulty. Vibration levels with the H-37 helicopter, to be used for the final test phase of the program, were expected to be much greater in amplitude and it was felt, at this time, that it would be necessary to isolate the entire sensor unit. This problem was discussed with engineers of the Lord Manufacturing Company of Erie, Pennsylvania, and appropriate isolation mounts were ordered for a new sensor mounting assembly.

Erratic operation of the control unit was found to be due to a malfunction of a monostable multivibrator along with induced pulses originating in the discharge current assembly.

The high voltage power supply performed quite well except for an occasional shift in DC level to the polarity control circuitry. This situation was found to be a further result of vibration effects.

Figure 13 is a record of discharge characteristics with the high voltage manually controlled to apply 25 KV continuously to the probe. The aircraft was flown at an altitude of 1500 feet directly over Niagara Falls at an airspeed of 80 knots indicated. The initial field indicated is on the order of +600 v/m at time of discharge voltage turn-on. Reduction to a near zero value was effected in about $2\frac{1}{2}$ seconds with a discharge current of approximately 7 microamperes. With the discharge voltage removed, the rate of charge buildup to a peak value of +800 v/m is on the order of 375 volts/meter/second, which when compared to the discharge rate of 250 volt/meter/second indicates the 25 KV applied is less than required to establish zero condition.

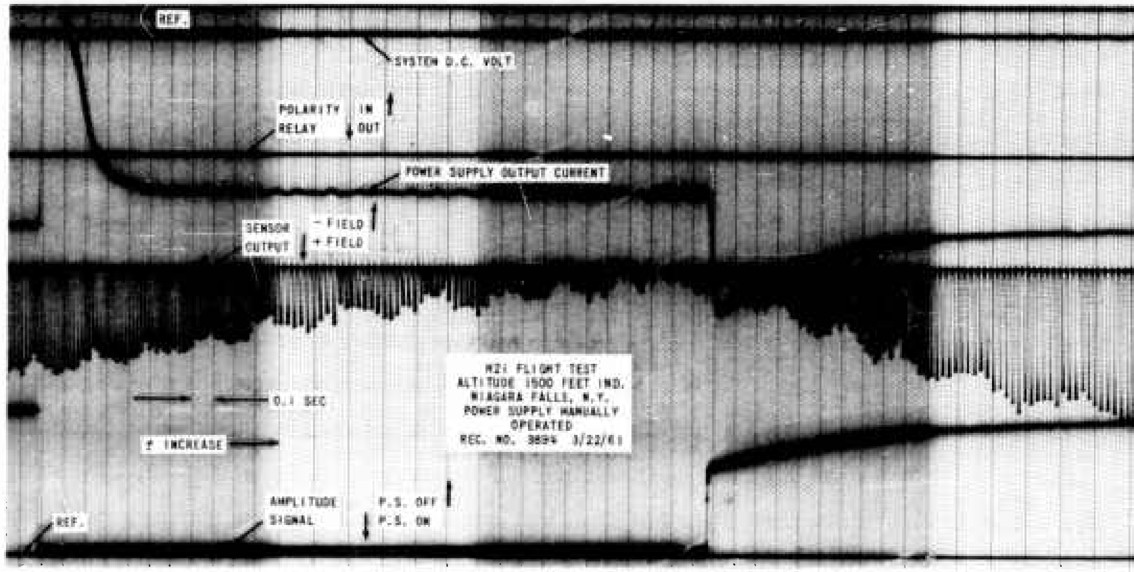

REF.
SYSTEM D.C. VOLT
POLARITY RELAY
IN
OUT
POWER SUPPLY OUTPUT CURRENT
SENSOR OUTPUT
- FIELD
+ FIELD
0.1 SEC
f INCREASE
MZi FLIGHT TEST
ALTITUDE 1500 FEET IND.
NIAGARA FALLS, N.Y.
POWER SUPPLY MANUALLY OPERATED
REC. NO. 3894 3/22/61
AMPLITUDE SIGNAL
P.S. OFF
P.S. ON
REF.

Figure 13

Figure 14 is a record of system operation in an automatic mode. Test conditions were the same as those for Figure 13 relative to altitude, air speed, etc. At time of turn-on, indicated field strength is on the order of +900 v/m. A 1.5 second hold delay is seen immediately after system turn-on is initiated during which time the control unit determines proper polarity for the sampled inputs. Reduction of the field to a system threshold level of 300 v/m was effected in less than 3 seconds. A discharge slope of approximately 200 volts/meter/second at $4\frac{1}{2}$ microampere average discharge current can be readily derived from this sequence. A wavering of the polarity control signal level can be seen, but amplitude variation is not sufficient to effect a polarity change.

Figure 15 is a plot of system operation for a negative field. For this test sequence, the system was made active before entry into the negative field region. Prior flights in this area for similar aircraft conditions (altitude and airspeed) indicated a peak field of approximately -1000 v/m was experienced. As the run was started, during the field transition through zero, system operation was properly quiescent. At approximately 250 v/m, the polarity relay was initiated and power supply turn-on effected. This immediate power supply turn-on indicated the hold monostable did not function at this point and was later repaired. Proper system action can be observed relative to pulse on-off rates required in order to maintain the average level of aircraft charge to the nominal threshold level.

Figures 16 and 17 are results of final laboratory tests conducted. With a positive field of 1500 v/m effected, normal system action can be seen in that a hold is initiated; polarity relay is in the off position, and an average field reduction to system threshold level is effected with an average discharge current of 4 microamperes. As the field is forced into the negative region to a 1500 v/m level, the polarity relay was actuated and the field reduced effectively to the negative threshold level.

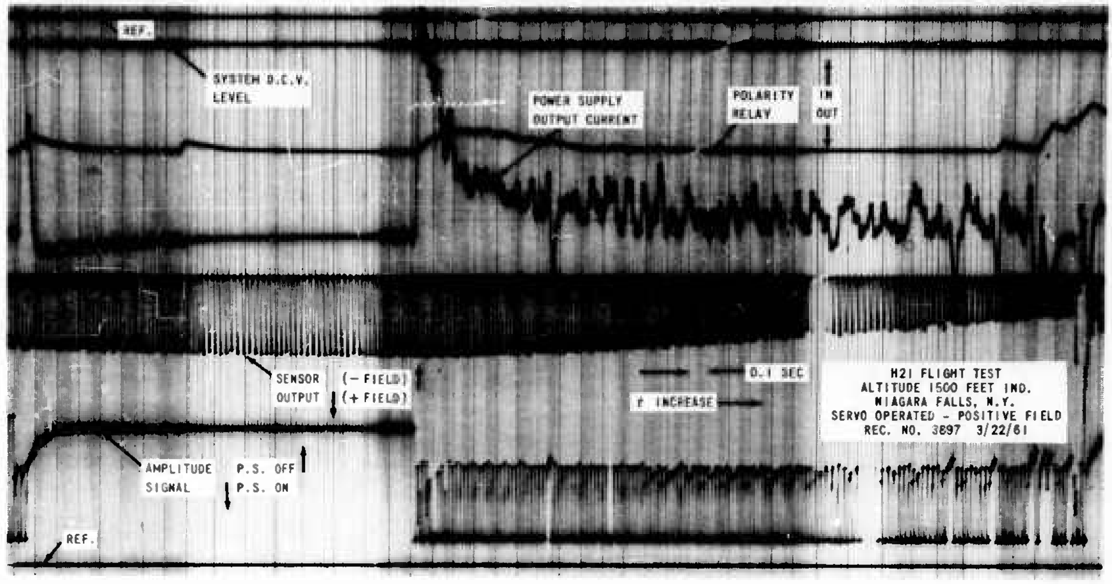


Figure 14

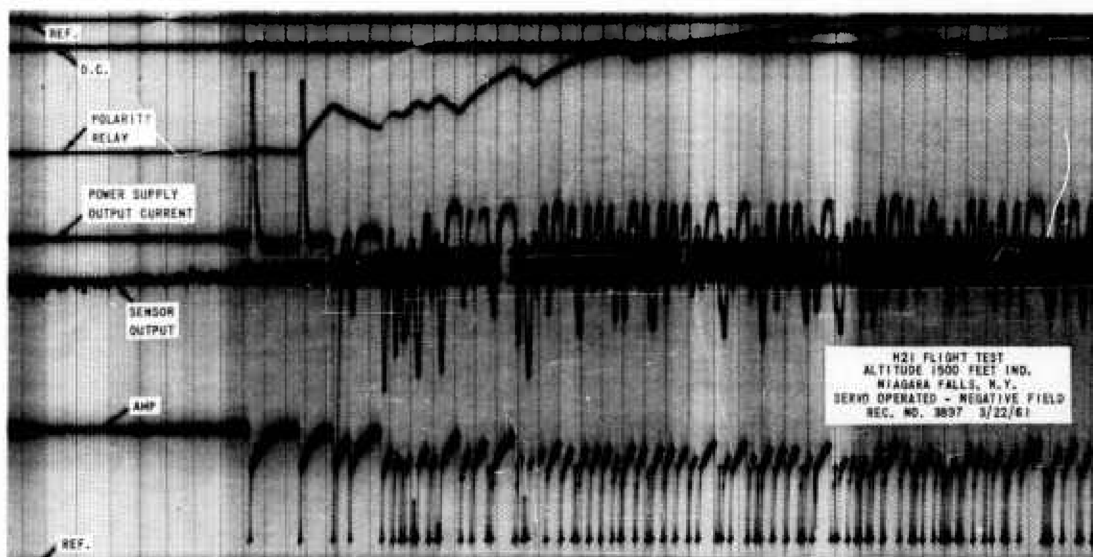


Figure 15

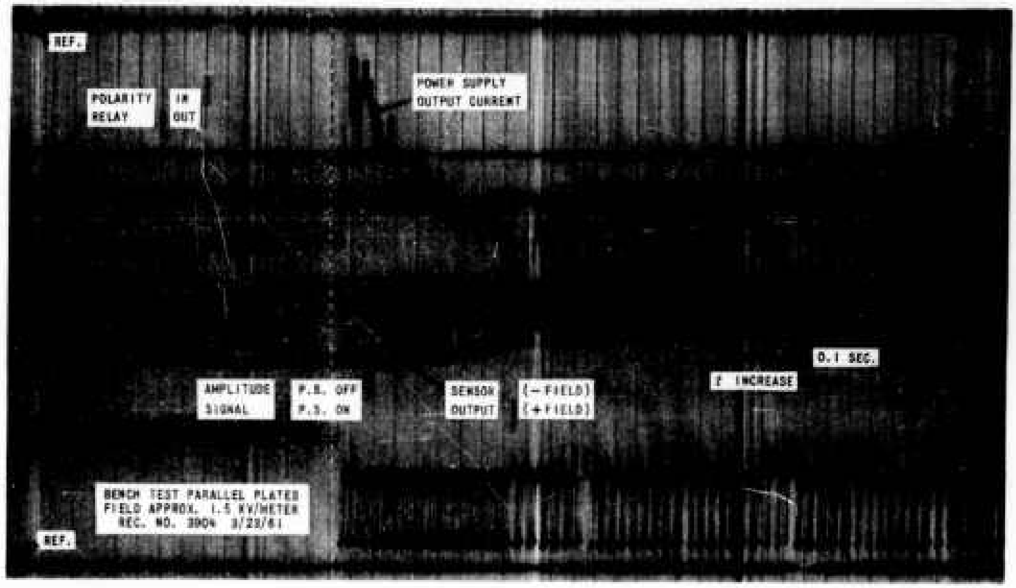


Figure 16

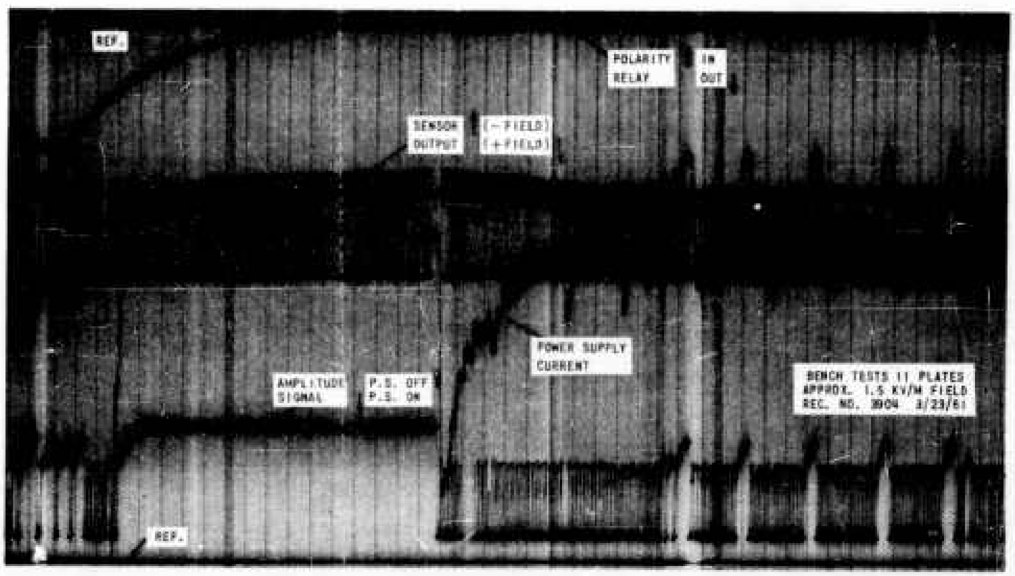


Figure 17

RESULTS AND CONCLUSIONS

The automatic system under test was designed on the basis of information derived from the previous feasibility program, with an H-37 used as a test aircraft at Edwards Air Force Base. System parameters such as time constants, charge levels, and corona discharge levels required for adequate reduction to a safe value were partial objectives of the feasibility program. Test data were, of necessity, relative and based on controlled precalibrated conditions.

In general, an H-37 aircraft, in a hover attitude at 25 feet was found to build up a charge voltage of approximately 50 KV. A discharge voltage on the order of 12 KV and discharge current of 2 microamperes were necessary to reduce this charge voltage to approximately zero.

During these preliminary flight tests, with the H-21 aircraft, it was found that 25 KV, continuously applied, with a discharge current of 7 microamperes, was necessary to reduce the aircraft charge to a near zero level (Figure 13). With the system operating in an automatic mode, this same condition prevailed with a discharge current of $4\frac{1}{2}$ microamperes over a longer period of time (Figure 14). For both of these sequences it was necessary to experience a reduced initial field by increasing aircraft altitude over the charged area. At altitudes less than 1500 feet, the aircraft induced charge was much greater than the sensor saturation level of 1000 v/m and a discharge current of 7 microamperes was not adequate to reduce the charge below this level during the time interval of flight over the charged area.

In order to obtain a feel for the actual aircraft potential relative to indicated field strength, the aircraft was hovered at 25 feet and the charge potential was measured to be +3 KV. Indicated field strength at this time was +200 v/m. With a similar sensing technique with an H-37 aircraft, a field strength of 1000 v/m was measured for a 3 KV potential buildup. These comparative figures point out that more investigative type work is necessary in order to establish new system parameters when aircraft other than the H-37 are considered. Induced fields experienced in this local area were much greater than system design capabilities, and test results should be analyzed with this in mind. Conditions as severe as these were not anticipated for the Phase III portion of the program.

It was concluded that system operation was compatible with design objectives and met requirements for the final test phase with the H-37 aircraft.

VII REMOTE BASE FLIGHT TESTS

A. INTRODUCTION

Arrangements were made, during the early part of the program, to conduct extensive field tests at Edwards Air Force Base, California. The fundamental purpose of these tests was to verify system performance for a hover attitude at 25 feet altitude, along with an investigation of system component operation for various aircraft attitudes.

The following basic parameters were recorded during flight tests with a CEC, type P-3, recording oscillograph:

1) Primary supply voltage
2) Sensor output signal
3) Amplitude control signal
4) Polarity control signal
5) Discharge current

Ambient temperature, relative humidity and barometric pressure were tabulated in addition to visual observation of corona discharge current.

B. AIRBORNE INSTALLATION

The automatic corona discharge system was installed in a Sikorsky H-37 rotary wing aircraft Serial No. 55621, at Edwards Air Force Base, California. Personnel of the Aviation Test Office, Transportation Materiel Command, effected the actual aircraft modification and equipment installation under CAL supervision, with this office supplying flight safety and aircraft limitation requirements and information.

The electrostatic field sensor was mounted on the underside of the aircraft along its centerline, at approximately station 130 (Figure 18). The mounting assembly was designed to provide adequate isolation to vibrations anticipated.

The corona discharge probe assembly was installed at the extreme lower rear portion of the tail section (Figure 19). The mount was designed for external installation using an existing aircraft rivet pattern along with suitable doublers to maintain the structural strength. An electrical conductor was routed internally from the probe assembly to the high voltage power unit.

The system equipment package, shown in Figure 20, was located on the left hand side of the aircraft at approximately station 145. Position of this equipment was not critical and final choice was made on the basis of convenience. The package consists of the high voltage power unit, amplitude and polarity control unit, corona discharge current assembly, control panel, and CEC recording oscillograph. A Tektronix Model 321 oscilloscope with its associated DC to AC converter was mounted alongside of the package assembly. Primary power for the entire system was the aircraft 28 volt supply.

C. PRELIMINARY FLIGHT TESTS

The first flight test was conducted May 10, 1961, to observe general system operation and capabilities for various flight attitudes. System parameters were observed visually and no oscillograph recordings made at this time.

In a 25 foot hover attitude over a concrete ramp, indicated aircraft field was positive at the saturated 1000 v/m level. With the system switched to automatic mode, the positive field was reduced to the threshold level in approximately 4 seconds. Rapid overshoots were experienced in both the positive and negative directions indicating random trigger signals were present.

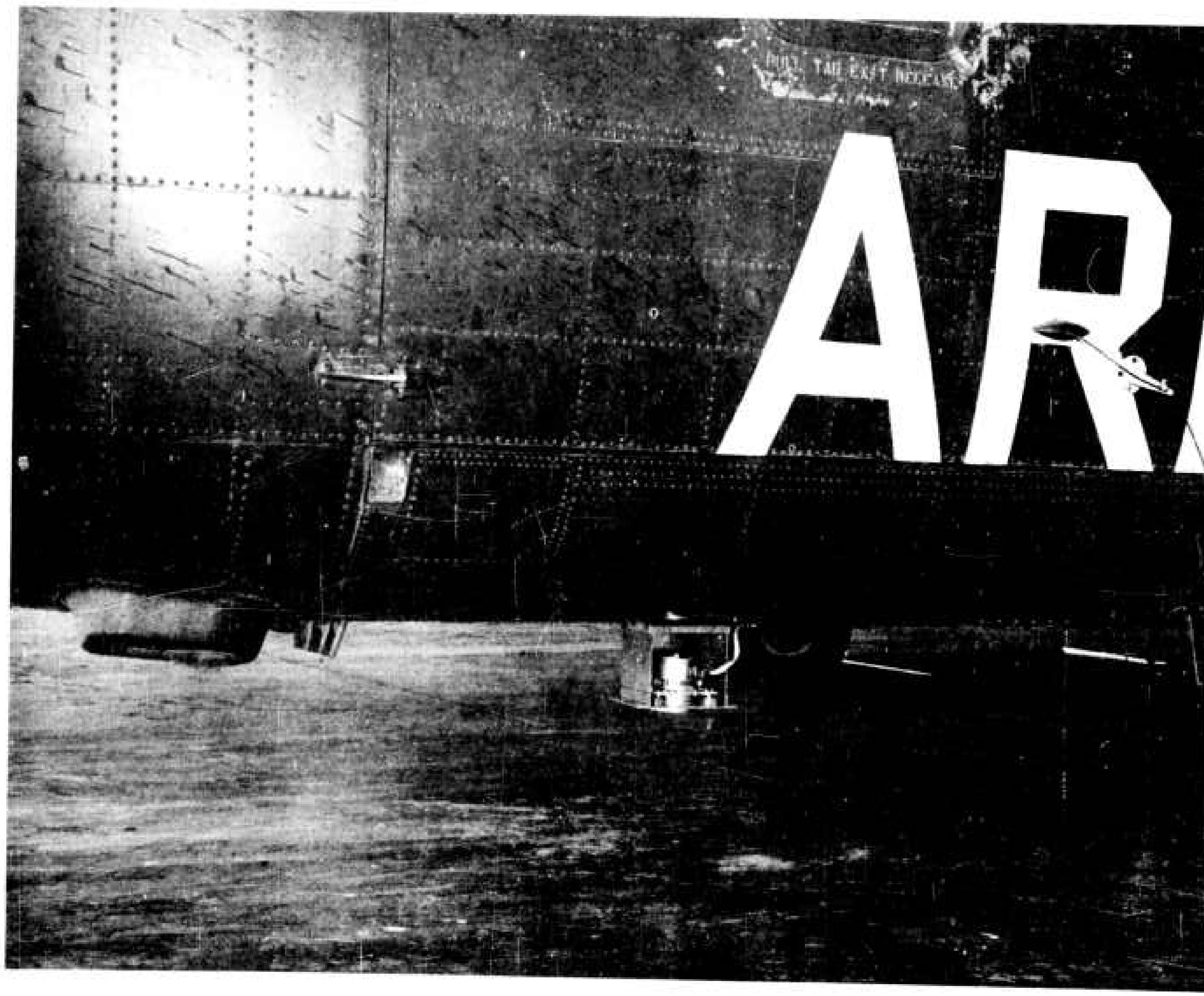


Figure 18 SENSOR INSTALLATION, H-37

Figure 19. Discharge Probe Installation, H-37.

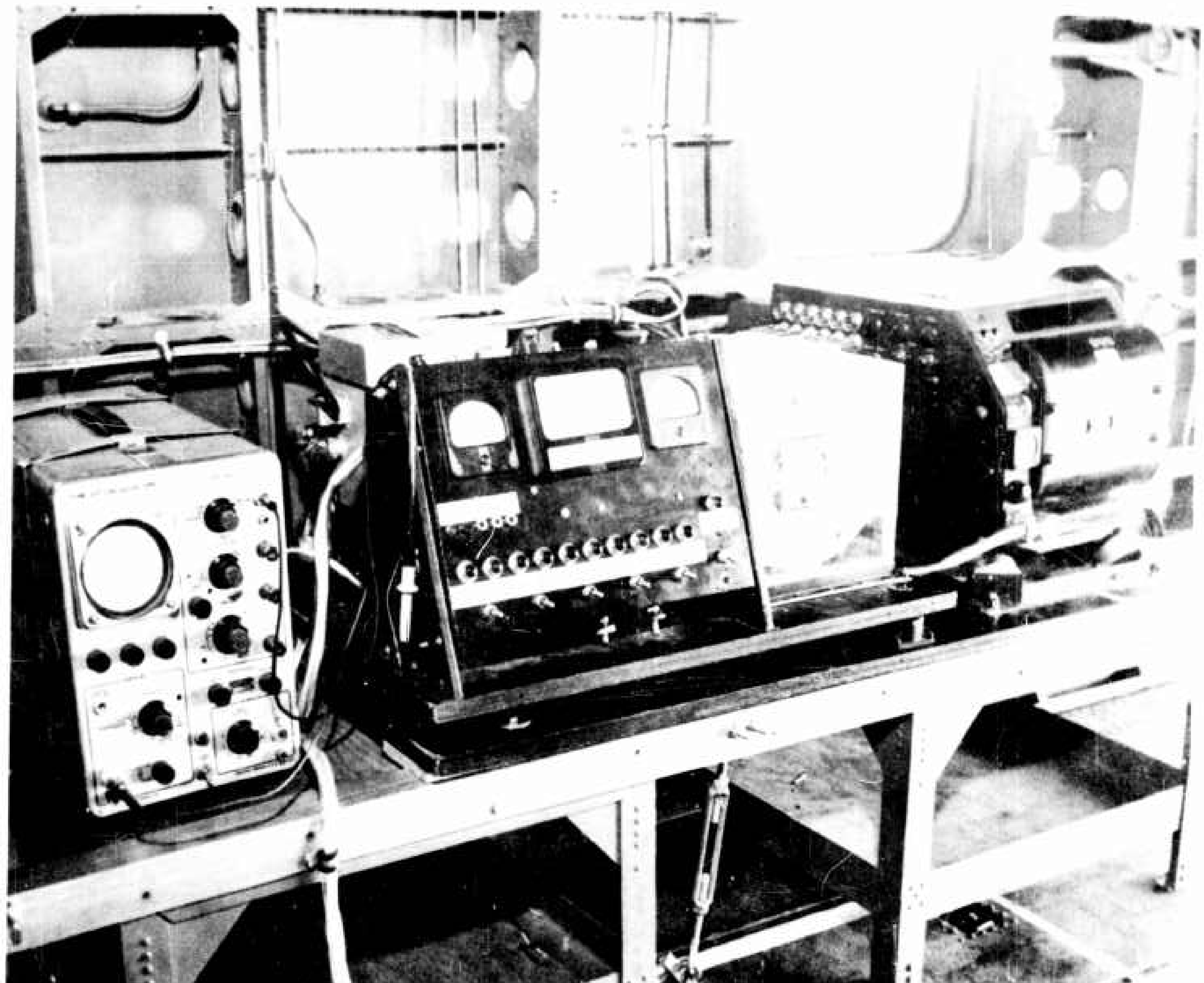


Figure 20. System Equipment Installation, H-37.

The shield cover was installed on the sensor and output signals observed on the Tektronix oscilloscope for various aircraft attitudes. In normal forward flight as well as in a stable hover, effects of aircraft vibrations were quite apparent. Peak sensor output voltage on the order of ±2 volts was common, increasing to more than twice this amount during aircraft transition from forward flight to hover. Since normal triggering levels are ±1.5 volts, the random triggering experienced during this flight was explained.

The basic vibration problem on the H-37 aircraft is very severe. Typical data for the average H-37 aircraft as obtained from Sikorsky engineering personnel are as follows:

3 cycles per second - amplitude ±.01 inch, lateral
15 cycles per second - amplitude ±.01 inch, vertical and lateral
50 cycles per second - amplitude ±.0015 inch, vertical

Vibration at the lower frequencies represents a major problem in that an appropriate isolation assembly with a resonant frequency less than 3 cycles per second would require elaborate design measures.

The fact that the only vibration sensitive portion of the sensor is the inherently microphonic electrometer tube indicated a more realistic approach to the problem would be to attempt more adequate isolation of this unit. By adding mass in the form of a lead shield around the tube and mounting on soft rubber, a combination was found which provided adequate isolation. Flight tests during the next few days verified that experimental results and system operation was normal, relative to vibration characteristics. However, vibration effects could be seen if aircraft transition from forward flight to hover was unusually "rough."

The last preliminary flight test was conducted May 18, 1961 in an ambient temperature of 84°F and relative humidity of 15%. System functions were observed for aircraft attitudes in forward flight, transition into hover, and hover. Results in general were good with aircraft charge reduced to and maintained within the threshold limits. Difficulty was experienced during the period of transition into hover, attributable to effects of vibration. In a hover attitude, a negative overshoot was experienced on the first cycle of operation only and the aircraft charge was then maintained below the threshold level in the positive region. With the aircraft flying at a normal cruise speed of 60 knots at an altitude of 1000 feet, overshoot, due to decreased relative capacity of the aircraft, was experienced for each cycle of operation. This mode proved useful for verification of bipolar capabilities of the system.

D. SENSOR MODIFICATION

During the early part of the flight test program, information was received regarding a new solid state device placed on the market by Crystalonics Inc., Cambridge, Massachusetts. The unit is a silicon, field effect transistor whose characteristics include: extremely high input impedance, low noise, wide temperature operating range and freedom from vibration and shock effects. Two type C-614 transistors were purchased and tested in various circuit configurations. Results indicated that the C-614 transistor could be used in the place of the electrometer tube .

Modifications to Sensor No. 2 were made, incorporating the C-614 in place of the electrometer tube. The transistor was rigidly mounted and circuit parameters designed for operation as a triode to take advantage of the low noise characteristics in this mode. Ample gain was derived which made further modification to other existing circuitry unnecessary. Comparative type tests were planned to evaluate the relative merits of this modification, as opposed to the electrometer tube configuration used in Sensor No. 1.

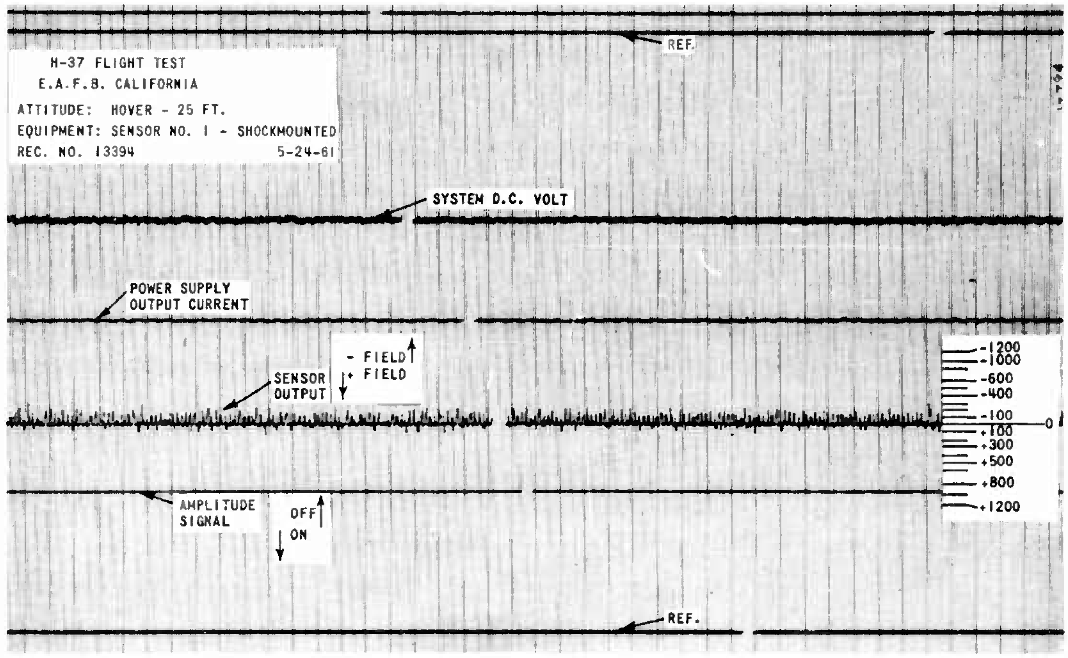


Figure 21

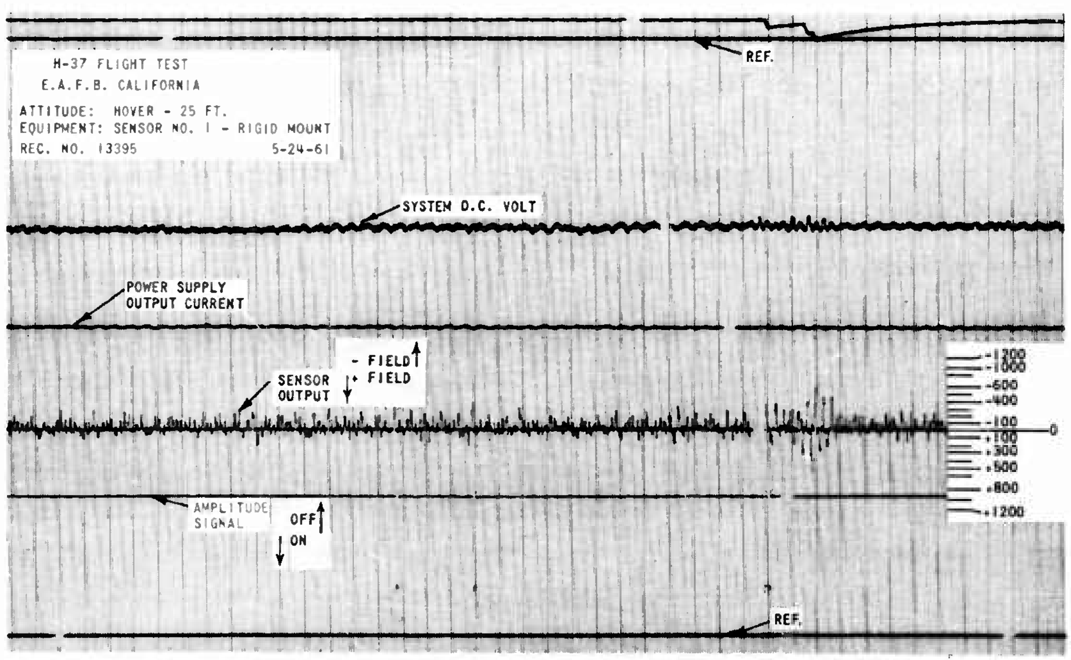


Figure 22

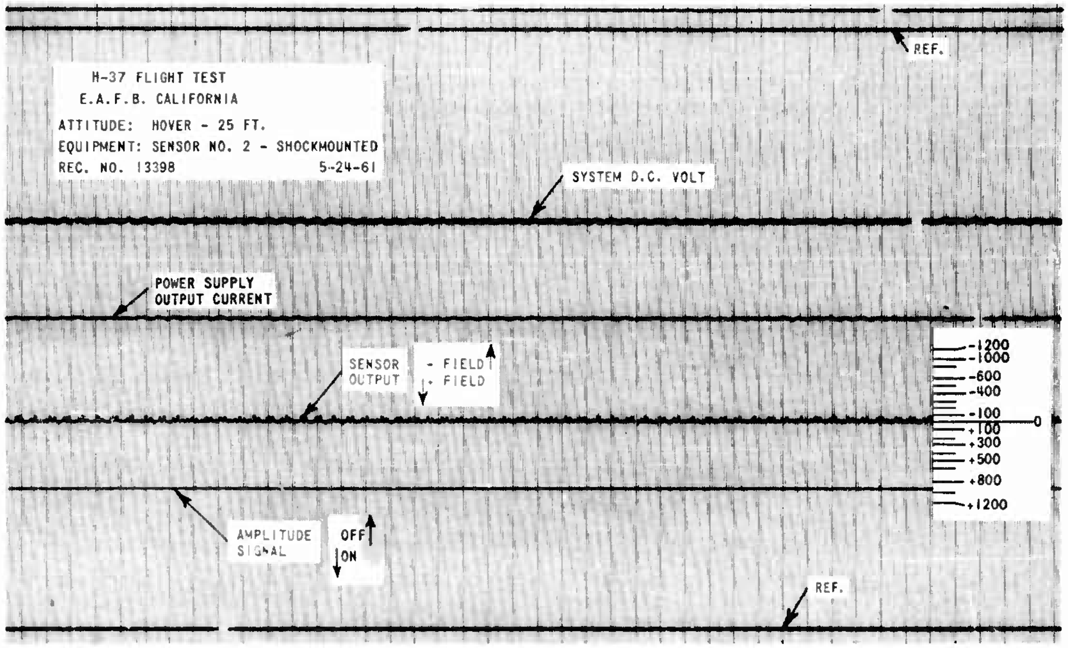


Figure 23

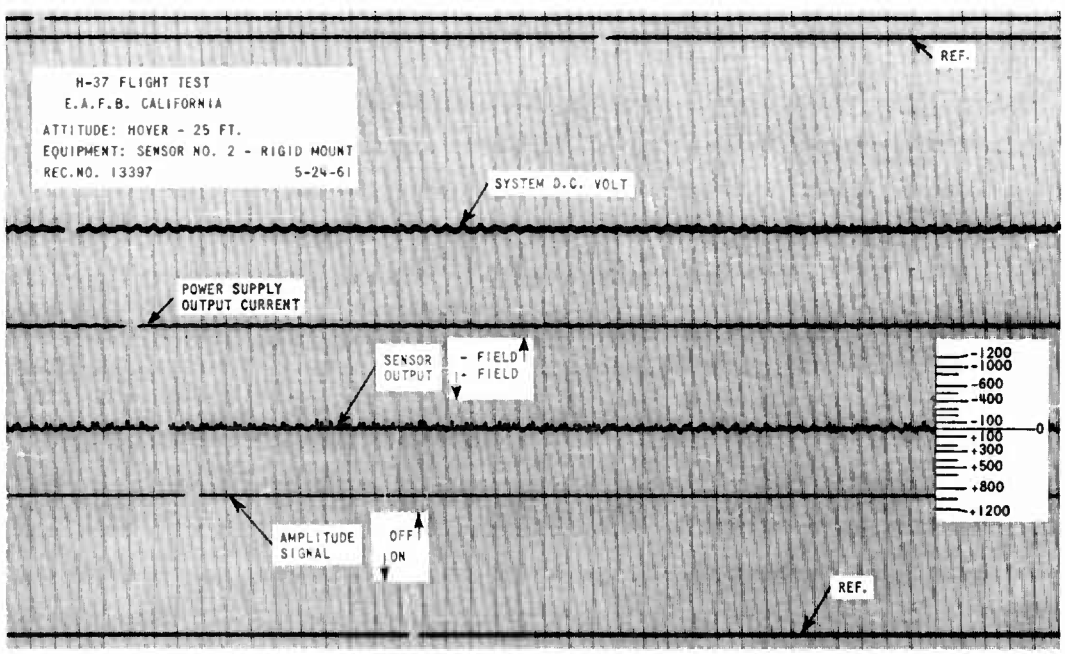


Figure 24

E. VIBRATION TESTS

In order to evaluate sensor operation with respect to aircraft vibration, a mount assembly was designed for internal, rigid installation on the aircraft floor at station 160. For all practical purposes, vibration characteristics at this point were the same as those existing at station 130 where the sensor is normally mounted. During the course of the in-flight test, both sensor No. 1 (electrometer tube front end) and sensor No. 2 (transistorized front end) were in turn operated in both a rigid and isolated mount configuration. Recorded results of system parameters for the aircraft in a 25 foot hover attitude appear in Figures 21 through 24.

With the shield cover installed, sensor output is negative and varies in amplitude due to inherent noise characteristics of the input device along with a small existing field due to contact potential etc. A variation of this initial signal is a major concern with vibration rather than its basic level.

Figure 21 is a record of operation results of Sensor No. 1 on the normal shock mount assembly. Output signal is stable and effects of vibration are almost non-existent. With the same unit rigidly mounted, effects of vibrations are seen throughout the run (Figure 22), and signal levels in three instances were large enough to trigger the amplitude control signal. Isolation of the electrometer stage is marginal for this mode of operation.

Figures 23 and 24 are runs with Sensor No. 2 shock mounted and rigidly mounted respectively. Both sets of data are clean and variations with respect to vibration barely perceptible. It can be clearly seen that transistor noise impedance is much lower than that of the electrometer tube, resulting in a much lower zero field output level.

In the interest of reducing wear and tear on integral components of the sensor, especially with regard to unusually hard landings, it was decided to incorporate the shock mount assembly as a final configuration. On this basis, it was concluded that both sensors were acceptable and compatible with system requirements. The transistorized version offers obvious advantages with regard to vibration and life expectancy. Both sensors were further tested in system configurations during the remainder of the field test period.

F. SYSTEM PERFORMANCE TESTS

During the remainder of the field work, systems performance tests were conducted including oscillographic recordings of system parameters for both forward flight and hover aircraft attitude. Calibration of system equipment for adequate coverage within anticipated system parameter excursions was effected using standard techniques. Detailed discussions of results are contained in the following paragraphs.

Flight Test - 5/26/61

Equipment compliment for this sequence consisted of system No. 1 components with the sensor unit installed on the shock mount assembly, along with complete instrumentation. Flight attitudes were varied from forward flight to hover at different altitudes.

Figure 25 is a partial record of system characteristics for a 25 foot hover over a concrete ramp. Equipment turn-on, with the exception of high voltage, was effected some time before the start of the run. Primary voltage was applied to the corona discharge supply at time T_o. Two or three amplitude control pulses effectively triggered high voltage circuitry before the monostable output level decreased below a relative "And" circuit threshold point. Graphically this can be shown as follows: reference to the control system block diagram is made throughout this discussion.

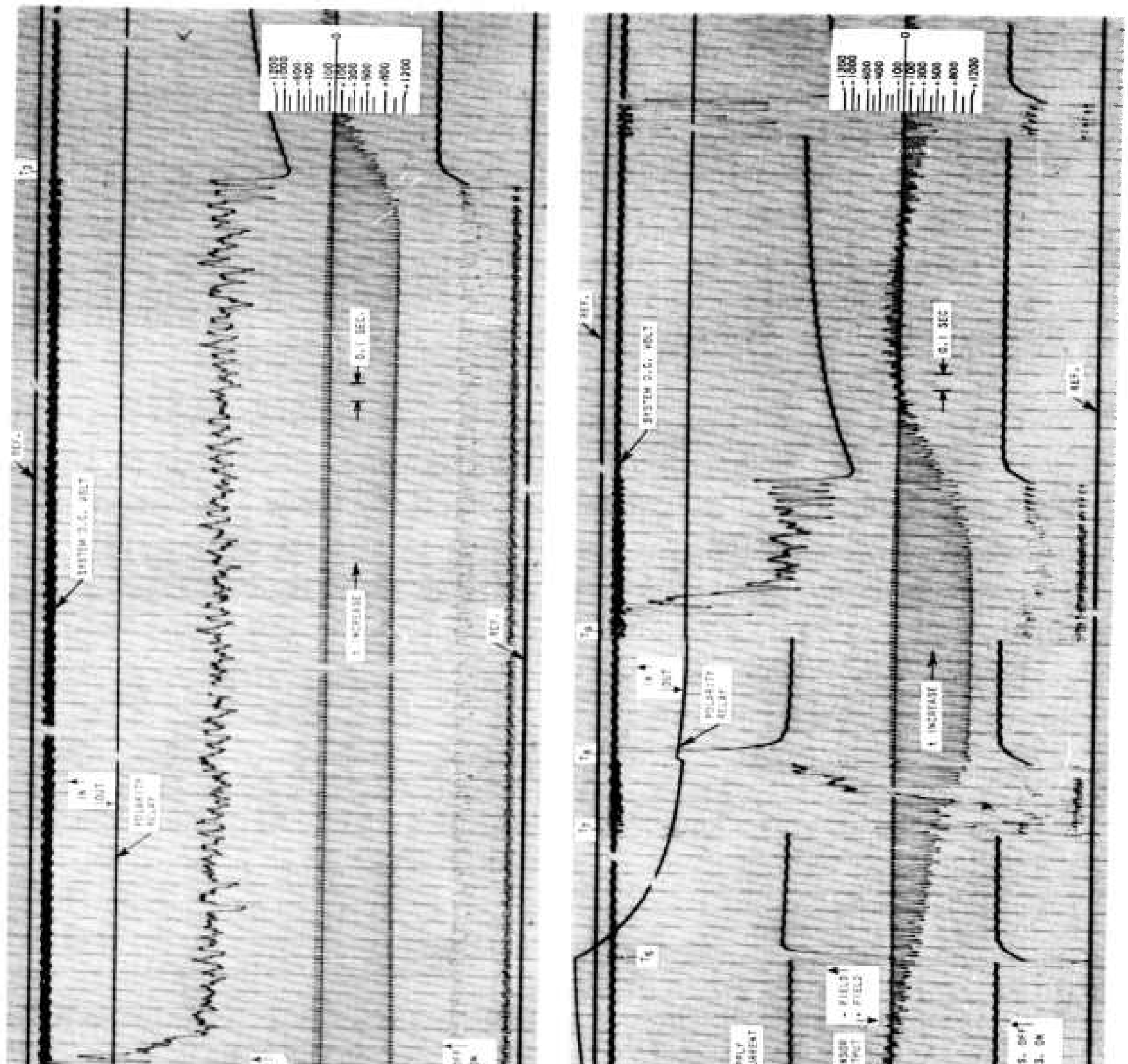
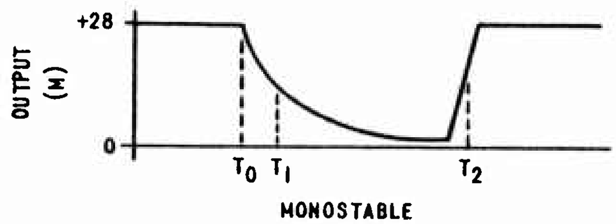

SYSTEM D.C. VOLT
POLARITY RELAY
POWER SUPPLY OUTPUT CURRENT
SENSOR OUTPUT
AMPLITUDE SIGNAL
INCREASE
0.1 SEC
REF.

Figure 25



$T_0 \longrightarrow T_1 \cong$ 30 MILLISECONDS
$T_1 \longrightarrow T_2 \cong$ 700 MILLISECONDS

At time T_O, inputs to A_1 (positive "And" circuitry) are present and proper. As seen from the record, polarity is positive and above the required threshold level (S); the polarity bistable (P) is providing a positive polarity signal and the monostable output (M) is normally high for a quiescent condition. With primary voltage applied to all circuits of the power supply simultaneously, effects of a transient condition are seen in the form of a slight deviation of the polarity signal level. It is believed that a large pulse, too high in frequency for the response of the oscillograph, is present at this time. Effective triggering of the monostable is caused and the "hold" monostable pulse initiated. The monostable output level to A_1 arrives at the "off" threshold level in approximately 30 milliseconds (T_1). During the interval to T_2 (700 milliseconds), A_1 output level is zero resulting in loss of driving signal to the amplitude bistable circuitry A. Application of boolean algebra to the above results in the following equations:

T_o	T_1	A_1	$= (S)(M)(P)$	$= (1)(1)(1)$	$= 1$
T_1	T_2	A_1	$= (S)(M')(P)$	$= (1)(0)(1)$	$= 0$
T_2		A_1	$= (S)(M)(P)$	$= (1)(1)(1)$	$= 1$

At T_2, the corona discharge system is activated at the full rate of 40 cps for a period of approximately 5 seconds. During this period a 22 KV pulse is applied to the discharge point and average discharge current is on the order of 5 to 7 microamperes. The initial field, which is much greater than the saturation level of 1000 v/m, is reduced at T_3 to the system threshold level and the amplitude control signal reduced to the off or zero level. The electrostatic field continually decreases through zero and into the negative region at a rate dependent upon aircraft charge characteristics.

On the second cycle of operation, an overshoot occurred which exceeded the negative threshold level and brought to light a malfunction of system operation under a particular set of conditions.

At T_4, the positive threshold level was reached and amplitude control of the discharge mechanism effected for about 300 milliseconds which was sufficient to force the field to less than the positive threshold point (T_5). At T_6, the negative overshoot is sufficient to initiate a high voltage polarity change. The monostable hold cycle is initiated as a result of this command and at T_4 plus 30 milliseconds, M becomes M'. During this interval, the polarity relay is off and a positive current spike is experienced. Energy content of this transient is not sufficient to effect a noticeable field strength change; therefore, although this action is undesirable, it is not detrimental to overall system performance and as such does not warrant added circuit complexity necessary for remedial measures.

During the next polarity reversal sequence a system malfunction occurred. At T_6, the positive threshold level was exceeded and both amplitude and polarity function time delays initiated. Control unit inputs at T_7 were correct for amplitude triggering since the polarity relay change command had been given and it was assumed that reversal would be effective within the monostable time delay period. Since the relay did not drop out until T_8, the power supply was actuated in opposite polarity increasing the positive incident field even further. Relay drop out occurred at T_8 and the resulting transient succeeded in initiating a monostable time delay. At T_9, signals were sampled by system logic circuitry and correct operation continued.

The remainder of the run contained no overshoots and system effectiveness displayed. Aircraft incident field was maintained well within the ±500 volts per meter design points.

System parameter recordings for altitudes of 1000 feet and 500 feet with the aircraft in forward flight further verified the system malfunction to be inherent. Due to reduced aircraft capacity in this mode, overshoots into the negative region occurred during each cycle of operation.

Flight records for the 50 foot hover attitude showed proper system operation in that no overshoots were present. A portion of this sequence recording appears in Figure 26. The high voltage system was turned on (T_0) at T_1 minus 4 seconds and system positive threshold reached at T_2, in approximately 6 seconds. Average discharge current for this period was on the order of 5-7 microamperes. At T_3, the positive threshold point was reached and high voltage pulsed on for 200 milliseconds to drive the incident field effectively back toward zero. For the remainder of the run, the field varied from zero to +600 v/m with high voltage being pulsed on at appropriately the 500 v/m point. This action assumed a periodic trend with a period of approximately 1.5 seconds and pulse on time of 200 to 400 milliseconds. During the entire sequence, the polarity control signal was stable and properly quiescent.

It should also be noted that the 28 volt primary aircraft supply contained approximately one volt of ripple which increases to 3 volts peak-to-peak with a system load of 3 amperes. The source impedance of this supply appears unusually high but no ill effects in system operation was apparent due to this factor.

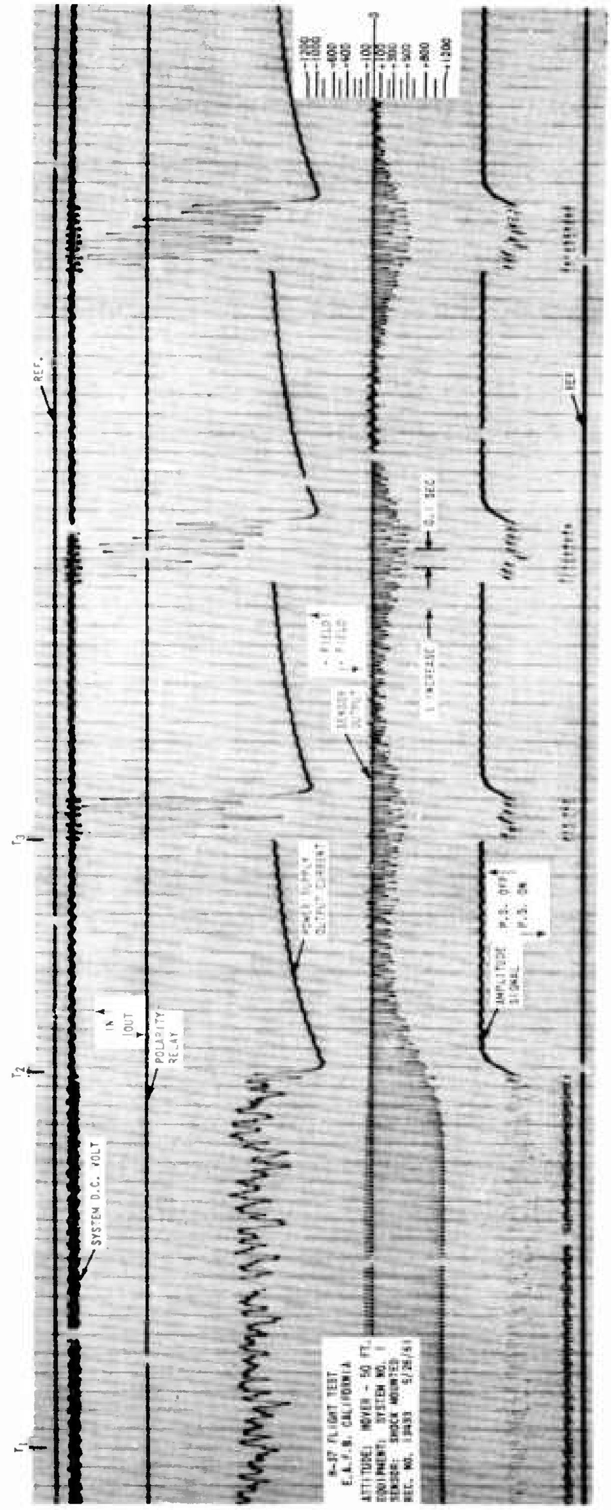

SYSTEM D.C. VOLT
POLARITY RELAY
IN
OUT
POWER SUPPLY OUTPUT CURRENT
SENSOR OUTPUT
AMPLITUDE SIGNAL
P.S. OFF
P.S. ON
REF.
0.1 SEC

Figure 26

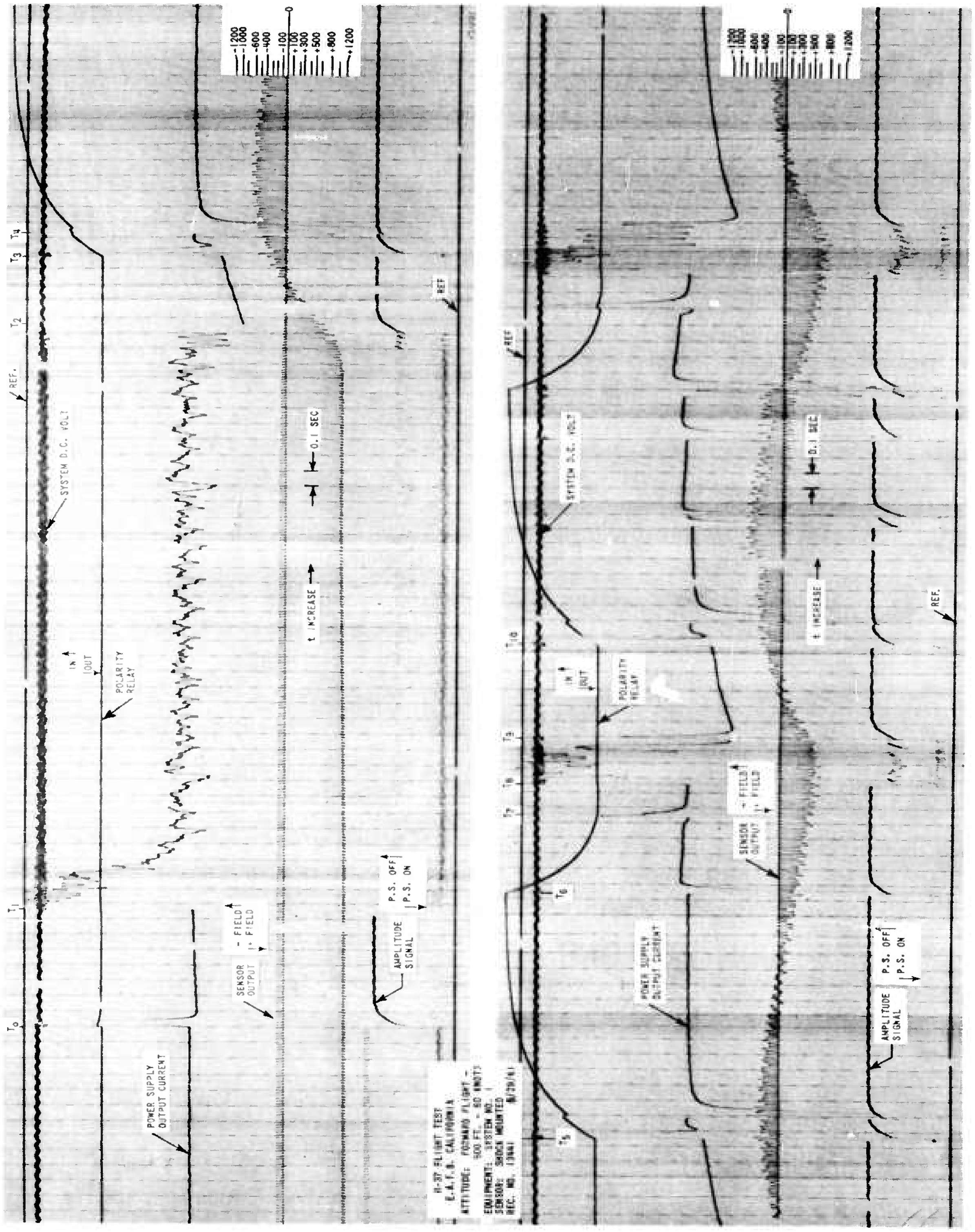
Flight Test - 5/29/61

In order to correct the system malfunction which occurs during a polarity reversal sequence, the polarity time delay circuitry was reworked to provide a shorter "off" time to insure drop out of the polarity reversal relay within the 700 millisecond monostable cycle. This remedial action was effective in the solution of this problem as seen in resulting flight test records.

Equipment compliment for the sequence was not changed from that used for the previous flight tests conducted May 26, 1961. Flight attitudes were also varied from forward flight to hover at different altitudes.

Figure 27 is a partial record of system characteristics for the aircraft in forward flight at 500 feet altitude and an indicated air speed of 60 knots. During the entire sequence, overshoots were apparent for almost every cycle of operation due to the decreased time constants present with the aircraft in this altitude.

As the run was started, aircraft charge was normally positive and well within the saturated region. System high voltage was applied at time T_o and system "hold" resulting from the monostable action was effective for 700 milliseconds. At T_1, high voltage appeared at the discharge point and was sustained until the positive threshold point was reached (T_2). System action was normally quiescent until the negative threshold was exceeded at T_3, at which time polarity reversal action was initiated. The polarity relay was closed at T_4 and monostable action initiated. During the next 700 millisecond interval the magnitude of the negative field was below the threshold point and normal field buildup toward the positive threshold occurred. It is interesting to note that time required to reduce the initial field to the 500 v/m level was approximately 3.8 seconds with a discharge current on the order of 5 to 7 microamperes. At T_5, a negative overshoot greater than the threshold level was experienced



POWER SUPPLY OUTPUT CURRENT
SYSTEM D.C. VOLT
POLARITY RELAY
IN
OUT
SENSOR OUTPUT
- FIELD
+ FIELD
AMPLITUDE SIGNAL
P.S. OFF
P.S. ON
t INCREASE
0.1 SEC
REF.

Figure 27

and polarity reversal initiated. Here again, the negative field remained below the trigger level after the normal monostable delay cycle and aircraft charge continued to decrease toward zero of its own accord. The positive threshold level was exceeded at T_6, and both polarity and amplitude time delays initiated. Polarity relay drop out occurred at T_7, with amplitude control circuitry causing high voltage pulsing at T_8. The sensed field was driven through the positive threshold resulting in system turn off at T_9. At T_{10}, the negative threshold level was exceeded and overall system action repeated. This sequence of operation displays the effectiveness of the remedial measures taken to correct the system malfunction experienced when a negative overshoot occurred during previous flight tests. The remainder of the run was repetitive with aircraft charge voltage maintained within design limits.

With the aircraft flying at 1000 feet at the same rate of indicated air speed, system action and discharge characteristics were found to be almost identical to those for the 500 feet altitude case.

Test results for both the 25 feet and 50 feet hover altitudes were also quite identical as was expected. Figure 28 is a recording of system action for an aircraft hover attitude at an altitude of 25 feet over a concrete ramp. Time required for reduction of the initial field to the system threshold level is approximately 5.9 seconds with 22 KV pulsed at a rate of 40 cps and discharge current on the order of 5 to 7 microamperes. Very little overshoot is experienced in the negative direction indicating that proper system time constants had been selected. Polarity control function is normally quiescent and effects of system transients are quite apparent in the 28 volt trace. The aircraft incident field was maintained between zero and +500 v/m for the remainder of the run.

Flight Test - 6/2/61

In order to evaluate performance of system No. 2, which differed from system No. 1 in sensor configuration, flight tests were conducted

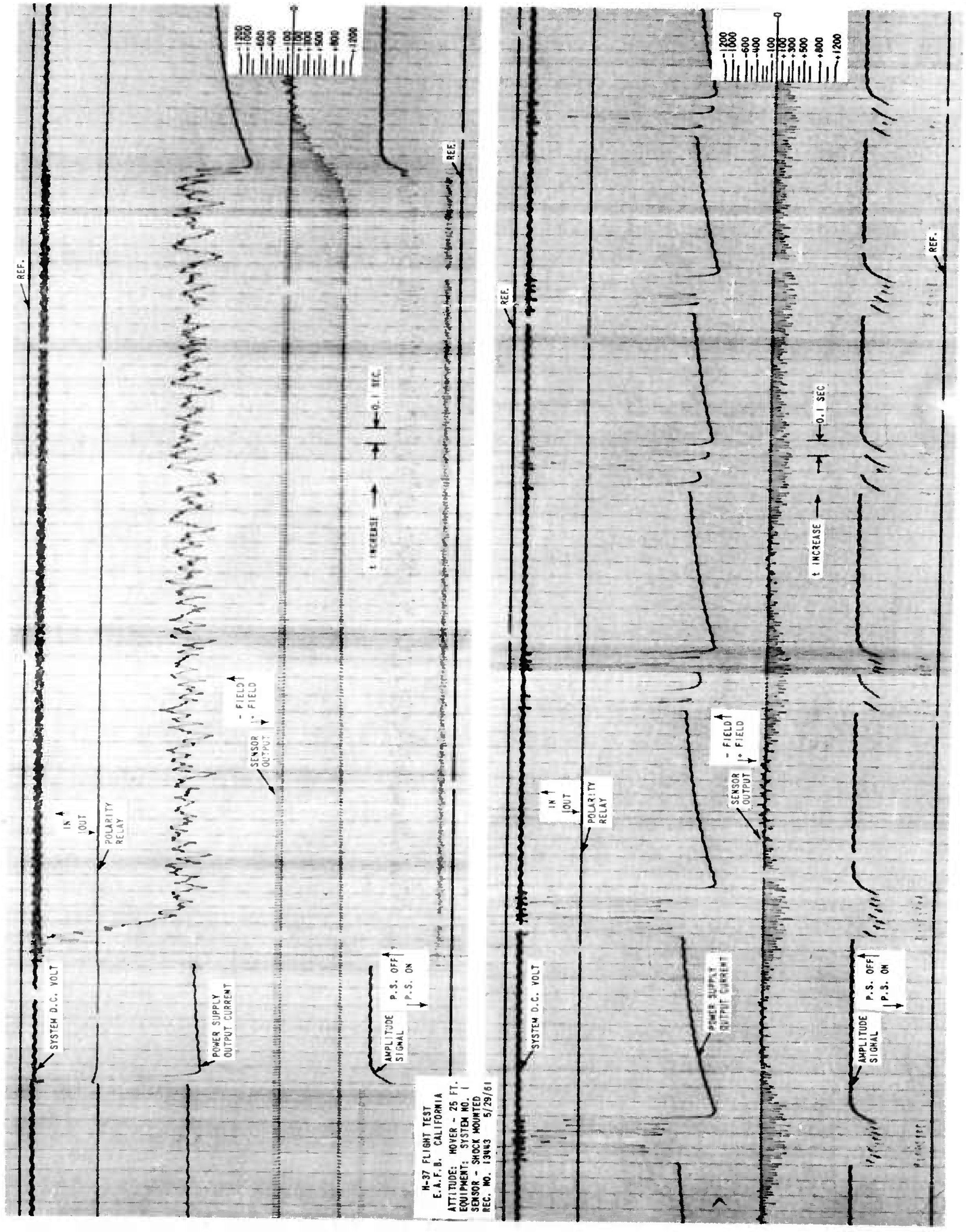

H-37 FLIGHT TEST
E.A.F.B. CALIFORNIA
ATTITUDE: HOVER - 25 FT.
EQUIPMENT: SYSTEM NO. 1
SENSOR - SHOCK MOUNTED
REC. NO. 13443 5/29/61
SYSTEM D.C. VOLT
POWER SUPPLY OUTPUT CURRENT
AMPLITUDE SIGNAL
P.S. OFF
P.S. ON
POLARITY RELAY
IN
OUT
SENSOR OUTPUT
- FIELD
+ FIELD
t INCREASE
0.1 SEC.
REF.
-1200
-1000
-600
-400
-100
+100
+300
+500
+800
+1200

Figure 28

at identical aircraft attitudes for comparative purposes. Runs were made with the sensor unit on isolators and rigidly mounted to the aircraft frame.

Figure 29 is a recording of system performance for a hover attitude at an altitude of 25 feet. The sensor unit, incorporating the C-614 solid state front end, was mounted on the normal shock mount assembly. Time required after application of high voltage to the discharge point to the positive threshold level was 4 seconds with an average discharge current of 5 to 7 microamperes. Overshoot into the negative region occurred on the first cycle of operation but with insufficient amplitude (-400 v/m) to effect a polarity change command. For the remainder of the run, aircraft sensed field was maintained between 0 and +500 v/m with stable system operation very apparent.

System recording of operation for the 50 foot hover case was also very satisfactory. Operation was similar to that for the 25 foot hover except for a decrease in the required time to reduce the initial field to the positive threshold level and for a more severe first overshoot. The decrease in time requirements (0.7 seconds) is very likely due to a slightly lower initial aircraft charge. Overshoot in the negative region was sufficient in amplitude to effect a polarity reversal which was not repeated throughout the remainder of the run.

System characteristics for the forward flight attitudes at altitudes of 500 feet and 1000 feet were also very similar to those for system No. 1. Time required for reduction of the initial field to the threshold level was similar, as was the action of polarity control circuitry, when required.

The sensor was rigidly mounted on the aircraft frame to observe operational characteristics under conditions of severe low frequency vibration. The standard flight attitudes were repeated and system recordings made. Figure 30 depicts operation in a 25 foot hover

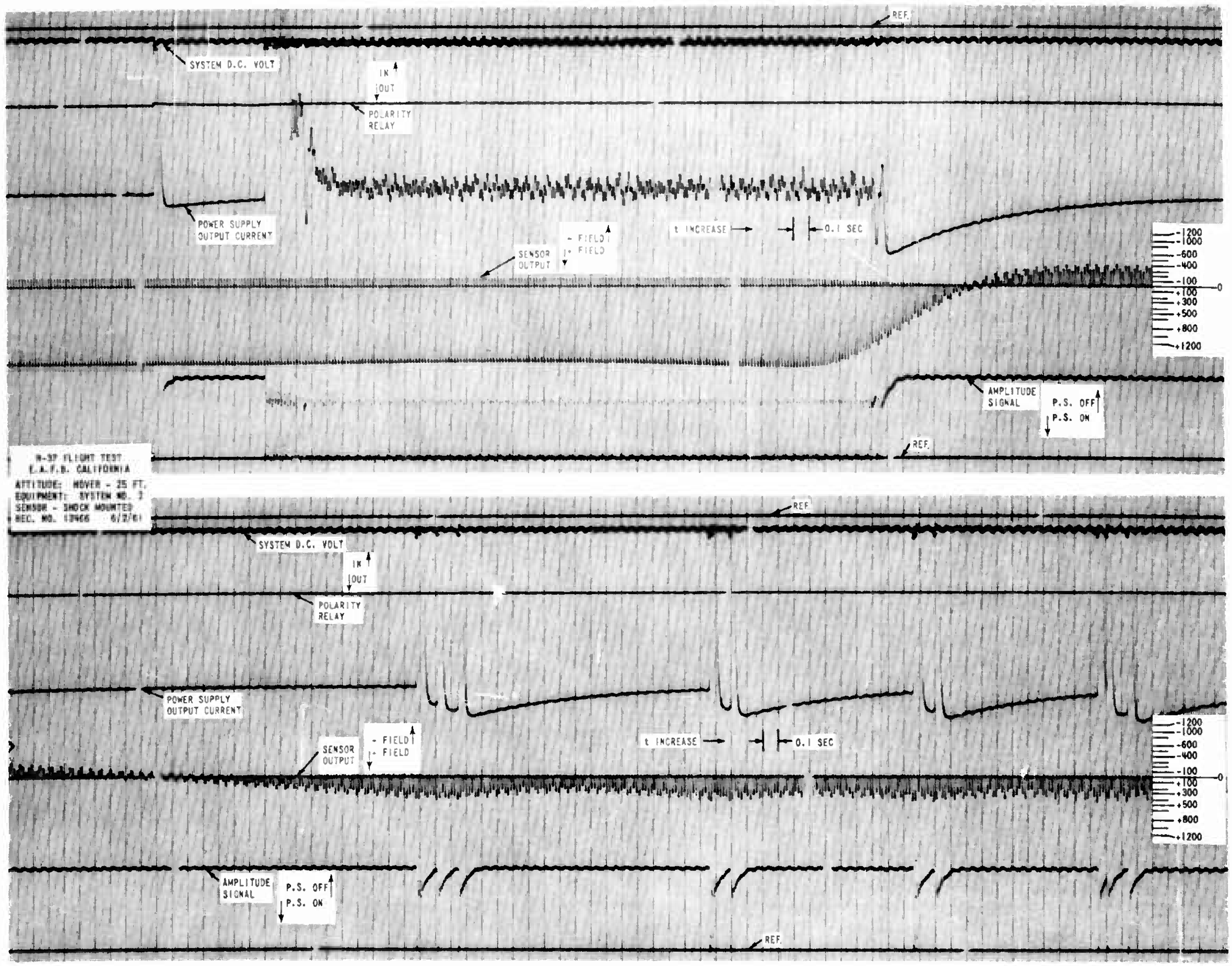

REF.
SYSTEM D.C. VOLT
IN
OUT
POLARITY RELAY
POWER SUPPLY OUTPUT CURRENT
SENSOR OUTPUT
- FIELD
+ FIELD
t INCREASE
0.1 SEC
-1200
-1000
-600
-400
-100
0
+100
+300
+500
+800
+1200
AMPLITUDE SIGNAL
P.S. OFF
P.S. ON
REF.
H-37 FLIGHT TEST
E.A.F.B. CALIFORNIA
ATTITUDE: HOVER - 25 FT.
EQUIPMENT: SYSTEM NO. 2
SENSOR - SHOCK MOUNTED
REC. NO. 13466 6/2/61
REF.
SYSTEM D.C. VOLT
IN
OUT
POLARITY RELAY
POWER SUPPLY OUTPUT CURRENT
SENSOR OUTPUT
- FIELD
+ FIELD
t INCREASE
0.1 SEC
AMPLITUDE SIGNAL
P.S. OFF
P.S. ON
REF.

Figure 29

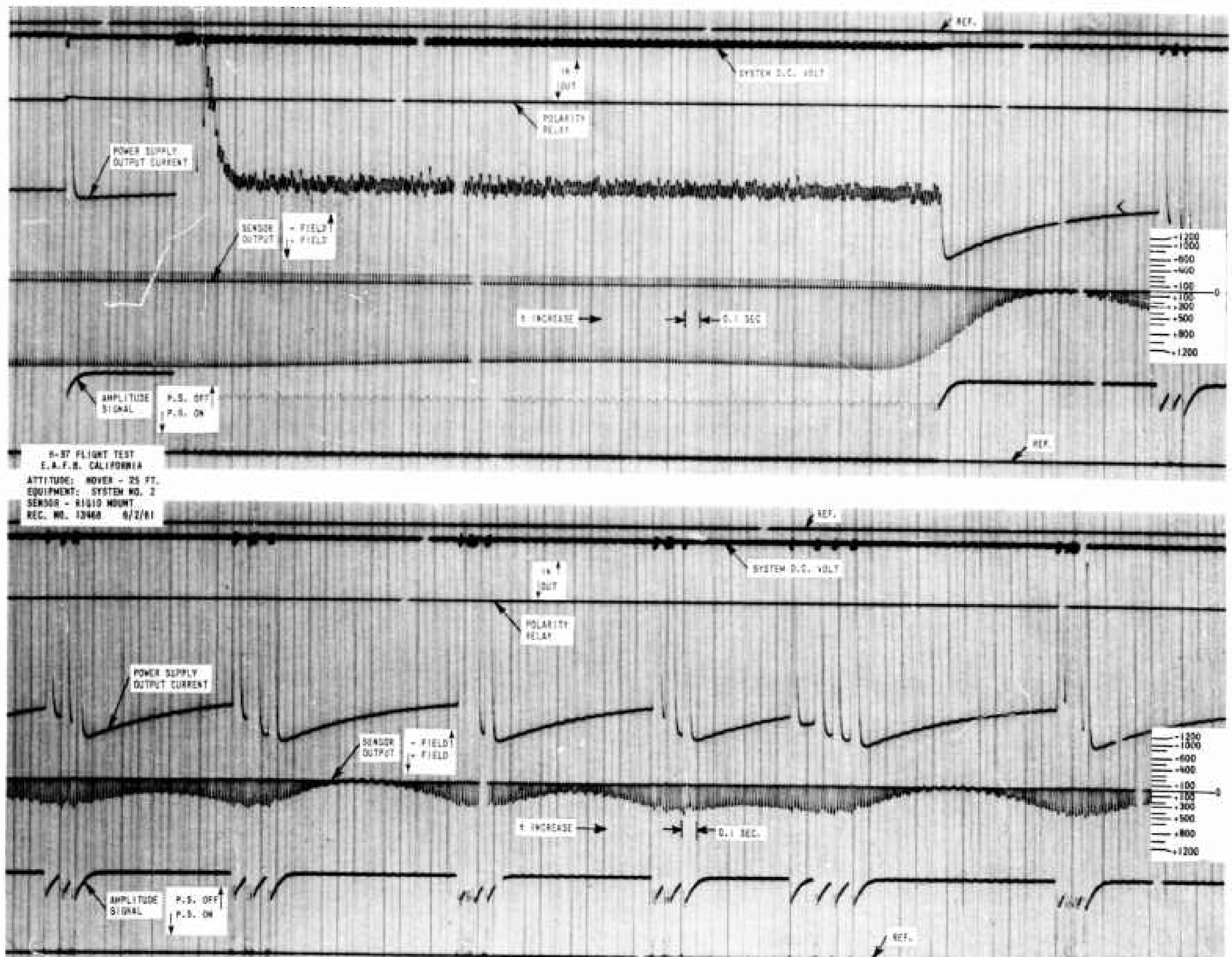

B-57 FLIGHT TEST
E.A.F.B. CALIFORNIA
ATTITUDE: HOVER - 25 FT.
EQUIPMENT: SYSTEM NO. 2
SENSOR - RIGID MOUNT
REC. NO. 13468 6/2/61
REF.
SYSTEM D.C. VOLT
IN
OUT
POLARITY RELAY
POWER SUPPLY OUTPUT CURRENT
SENSOR OUTPUT
- FIELD
+ FIELD
ε INCREASE
0.1 SEC
AMPLITUDE SIGNAL
P.S. OFF
P.S. ON
-1200
-1000
-600
-400
-100
0
+100
+300
+500
+800
+1200

Figure 30

attitude. Operation of all system components is seen to be normal with effective reduction to threshold level in 4.9 seconds. Overshoots were practically non-existent and extremely stable operation very apparent. A careful examination of the run in the reduced field area shows no apparent stray pulses due to aircraft vibration. Operation in the 50 foot hover, and in forward flight at increased altitudes, displayed normal sequences and absolute freedom from vibration effects.

In order to observe system performance with the aircraft capacity increased considerably from that experienced in a 25 foot hover, a record was made with the aircraft in a landing approach attitude. The run was started on the landing approach and continued through a sustained hover at an altitude of less than 10 feet. Aircraft capacity in this mode is greater than 1000 micro-microfarads.

Figure 31 is a recorded presentation of system operation for the above condition. High voltage was applied to the corona point 5 seconds prior to T_1, at which time the aircraft field was reduced to the positive threshold level. Discharge current during this interval was on the order of 5 to 7 microamperes. During the period T_1 to T_2, the aircraft was still in an approach attitude which results in a varying incident field. Results are seen in the form of an increased rate of high voltage firing at approximately a $2\frac{1}{2}$ cycle rate. A negative overshoot is experienced at T_2 and both polarity and amplitude delay circuitry properly actuated. During this delay period, the indicated field increased in the positive direction of its own accord, therefore high voltage actuation in the negative mode did not occur. Positive threshold level was exceeded at T_3 and an appropriate sequence of events occurred. Transition into hover, which is an extreme period of vibration took place some time between T_1 and T_3. It is interesting to note that no abnormal operational effects are detectable.


H-37 FLIGHT TEST
E.A.F.B. CALIFORNIA
ATTITUDE: HOVER - 10 FT.
EQUIPMENT: SYSTEM NO. 2
SENSOR - RIGID MOUNT
REC. NO. 13472 6/2/61
POWER SUPPLY
OUTPUT CURRENT
SYSTEM D.C. VOLT
POLARITY
RELAY

Figure 31

The remainder of the run was typical in operation with the helicopter sensed field maintained between zero and +500 v/m.

Flight Test - 6/2/61

In order to determine the voltage remaining on the aircraft for a nulled condition, the following procedure was used. The output of a Keithley high input impedance vacuum tube voltmeter, with a 1000:1 probe, was recorded on an oscillograph channel calibrated for a basic 0 to ±1000 volt range. An insulated conductor, with a 3 pound conductive weight attached, was used in conjunction with the Keithley to measure aircraft potential to ground.

The aircraft hovered at an altitude of 25 feet, directly over a steel mesh assembly covering an area of 64 square feet. The mesh was adequately weighted down to overcome the effects of rotor downwash and electrically connected to a standard static ground. The drop line was hand lowered until the conductive weight made contact with the steel mesh. Figure 32 is a recording for this attitude with the aircraft charge maintained within system threshold limits. The resulting aircraft to ground charge voltage is approximately +290 volts which is well below the design limit. Figure 33 is a photograph of the H-37 aircraft, taken during this test sequence.

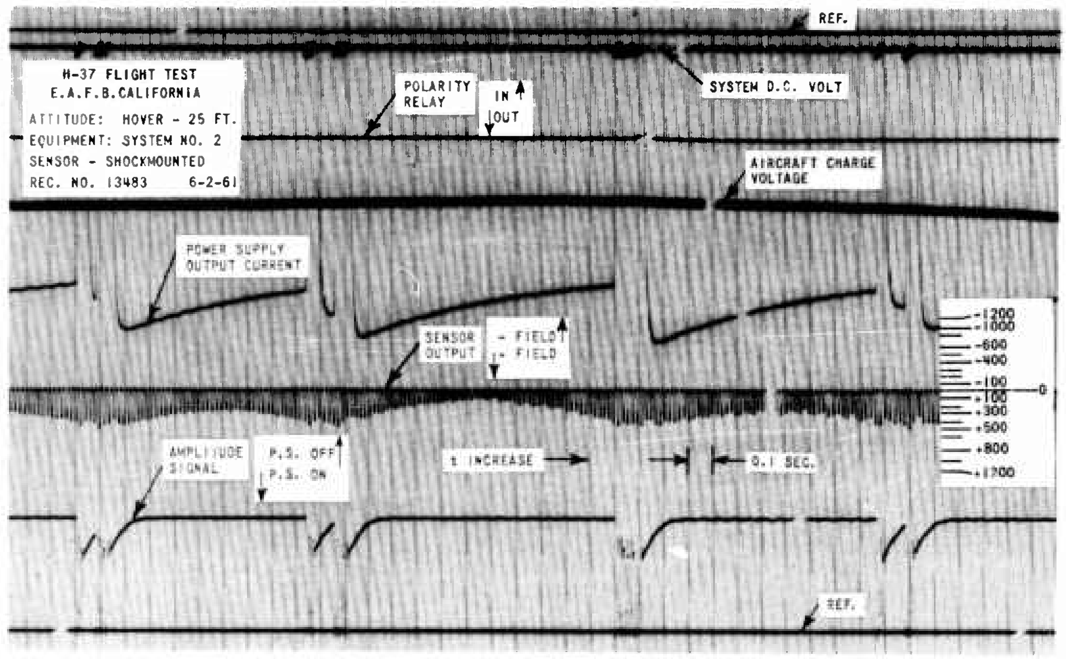

REF.
H-37 FLIGHT TEST
E.A.F.B.CALIFORNIA
ATTITUDE: HOVER - 25 FT.
EQUIPMENT: SYSTEM NO. 2
SENSOR - SHOCKMOUNTED
REC. NO. 13483 6-2-61
POLARITY RELAY
IN
OUT
SYSTEM D.C. VOLT
AIRCRAFT CHARGE VOLTAGE
POWER SUPPLY OUTPUT CURRENT
SENSOR OUTPUT
- FIELD
+ FIELD
-1200
-1000
-600
-400
-100
0
+100
+300
+500
+800
+1700
AMPLITUDE SIGNAL
P.S. OFF
P.S. ON
t INCREASE
0.1 SEC.
REF.

Figure 32

Figure 33. H-37, Hover Attitude.

VIII PHASE IV

A modification to the basic contract was made during the early part of the Phase II portion of this program, initiating a requirement for fabrication of a duplicate set of prototype discharge equipment. Installation was to be made in an H-37 aircraft at Ft. Eustis, Virginia, with appropriate flight tests to insure proper operation.

Effort was started on this phase in February 1961, and completed within a four month period. Initial bench tests were conducted at EAFB, California and operation found to be satisfactory. On the basis of test results with System No. 1, modifications were made to Sensor No. 2, to incorporate a solid state first stage, replacing the electrometer vacuum tube. Both vibration and complete system performance tests were performed during the Phase III portion of the program and reported upon in that section of this report. Results in general were very satisfactory as seen by reference to system performance recordings.

Upon completion of the field test program, Phase IV system equipment was repackaged with the incorporation of appropriate switching circuits and a field strength indicating meter. Total weight of this system package including all necessary cabling was 49.25 pounds.

Delivery to Ft. Eustis, Virginia, was made on July 12, 1961. Mr. S. B. Poteate, project officer on this program, made all arrangements for personnel needed to install the system on an H-37 aircraft. Preliminary system checks were completed and operation found to be normal.

A flight test was made July 15, 1961, in an ambient of 90°F, with relative humidity on the order of 90%. System operation was observed for a forward flight attitude at 1000 feet where bipolar capabilities were nicely displayed due to normal excessive overshoot.

At a 25 foot hover, the aircraft "zeroed" in approximately 4 seconds and remained within the threshold limits with triggering on the positive threshold only. Aircraft net charge was positive with respect to ground at all times.

IX PHASE V

The electrostatic field sensor was designed and developed for installation in an H-37 aircraft to measure aircraft charge and present this information in a manner compatible with an automatic discharge system configuration. In order to meet TRECOM requirements for a basic measuring device with extended ranges to ± 100, 000 volts per meter, a redesign of the basic unit was deemed necessary. This additional work was done on a modification to the basic contract.

A. SYSTEM

The measuring system is comprised to two units; a generating voltmeter for sensing magnitude and polarity of incident electrostatic field and a metering and control panel which includes switching functions as well as field strength indication. Power requirements are 28 volts DC at approximately 0. 5 ampere. Range of measurement is 0 to ± 100, 000 volts per meter in six steps.

A schematic diagram of the sensor electronics appears in Figure 34. The signal amplifier consists of four transistorized stages with an overall gain of approximately 500 at 40 cps. Input impedance on the order of 40×10^6 ohms is achieved through use of a field effect transistor (C-614) operating in a triode mode. Gain of this stage is approximately 4 with inherent noise down to an extremely low level. The following two stages are standard common emitter amplifiers. A signal level control, R12, is used for preventing overloading of Q_1 at maximum signal levels. A relatively low output impedance, along with adequate isolation,is achieved through use of an emitter follower output stage, Q_3. Voltage to the electronics is regulated

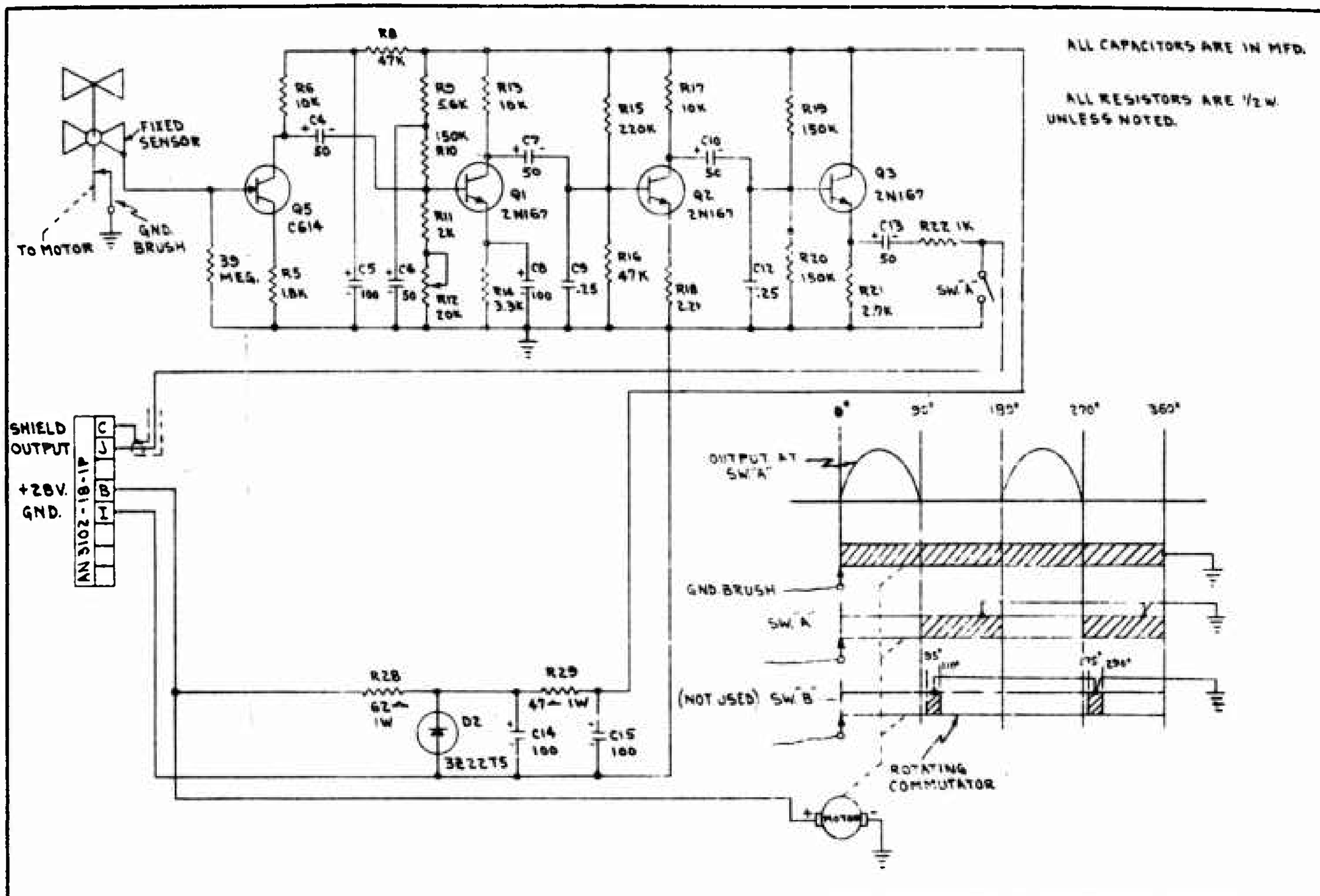


Figure 34 SENSOR SCHEMATIC (STATIC CHARGE)

at 22 volts which allows for primary voltage fluctuations of ±2 volts without effecting gain characteristics.

A schematic representation of commutator action also appears on this drawing. One commutator segment is used to maintain the shaft and rotor at case potential. The other, "SW_1A," provides a reference ground to the amplifier output through the current limiting resistor R_{22} for 90°; twice each revolution. The resulting half sine wave signal is shown for a sensed positive field.

A schematic diagram of the indicator and control assembly appears in Figure 35. A zero center, 50 μ a full scale meter, is used as the integrating device along with suitable series resistors for various scale factor attenuations for relative scale factors. A photograph of the unit appears in Figure 36.

B. OPERATION

The basic scale of this instrument, with the range selector switch in the X1 position, is 0 to ±1000 volts per meter. The range selector provides for electrical attenuation to extend this range by factors of 2, 5 and 10, for a maximum field strength reading of ±10,000 volts per meter. Two mechanical attenuators are supplied to provide further attenuation factors of 5 and 10, thus extending the maximum range to ±50,000 and ±100,000 volts per meter respectively.

Operation of the measuring device is simple and straightforward. With the range selector switch in the X10 position, power is applied by actuation of the "on-off" switch. A period of 30 seconds should be allowed for stabilization. The range selector is then switched to the proper position for adequate read-out. Field strength reading is made on the basic 0 to ±1000 scale and multiplied by the attenuation factors in use to arrive at indicated volts per meter.

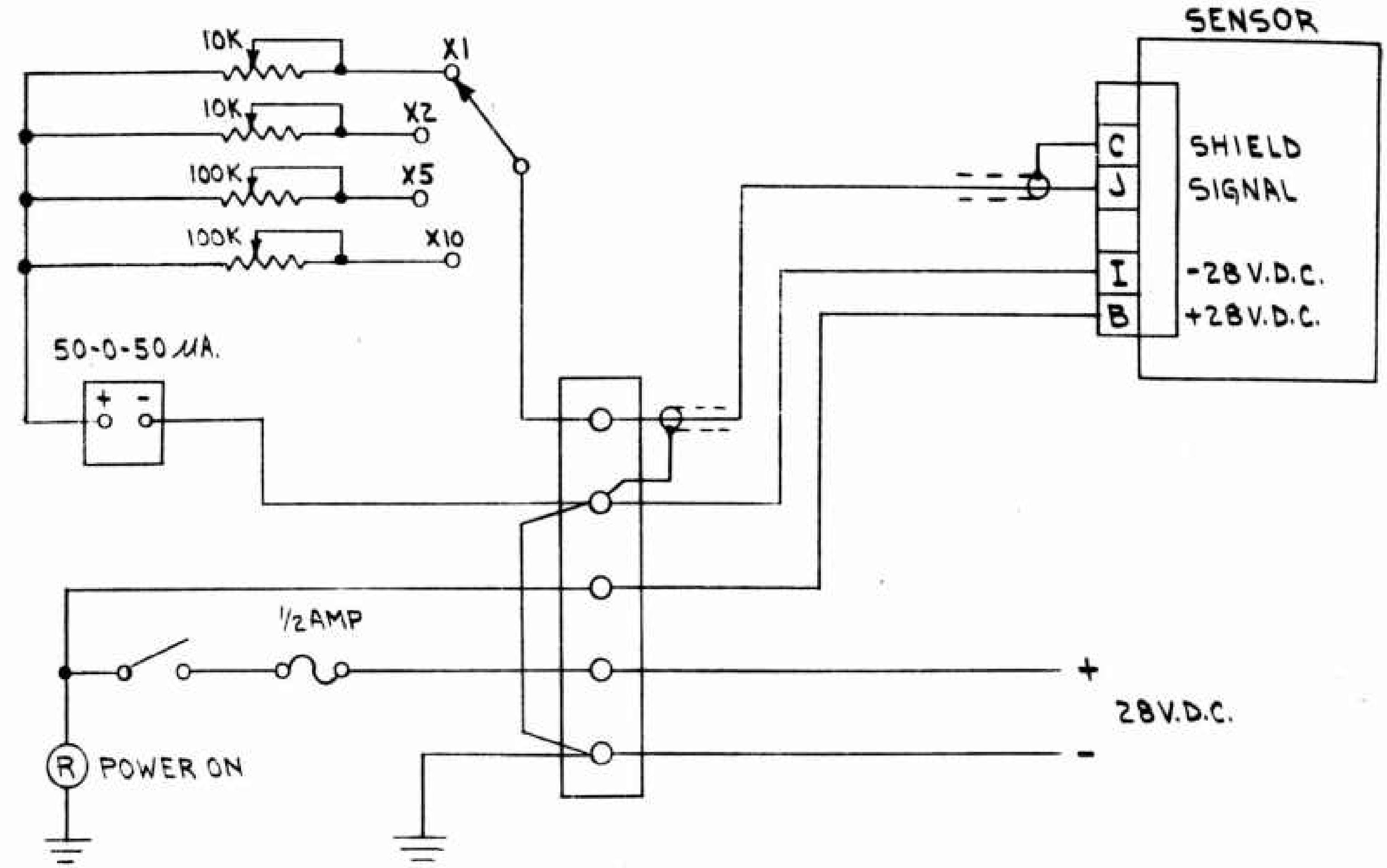


Figure 35 INDICATOR UNIT (STATIC CHARGE SENSOR)

Figure 36 ELECTROSTATIC FIELD SENSOR

C. FLIGHT TESTS

Delivery of this equipment was made to Edwards Air Force Base, California, August 8, 1961, and installed in the same H-37 aircraft used for the Phase III portion of the program. The sensor unit was shock mounted on the underside of the aircraft along its centerline at station 130. The metering and control panel was located on the left side of the aircraft at station 145, 48 inches above the aircraft floor with primary power supplied by the basic 28 volt aircraft system.

A total of three flight tests were made and charge buildup recorded for various aircraft attitudes. Ambient temperature varied from 84°F to 98°F; relative humidity from 17% to 35%, for the data runs.

D. RESULTS

Comparison of field strength data with results obtained during the feasibility program (Reference 2) brings to light an appreciable discrepancy. Average field strength readings for the hover attitude were on the order of 1/3 those measured with this device. These differences are attributed to a slightly different calibration setup, a change in dimension of the sensing plate and aircraft frame, and a different set of form factors resulting from a redesign of the original sensor configuration.

These field strength differences are not serious inasmuch as they are relative values and it is only important that a relationship to the aircraft charge in volts for a particular attitude be established. In order to do this, the aircraft was hovered at 25 foot and its potential to ground was measured with a high input impedance vacuum tube voltmeter and recorded along with its associated field strength reading. Unfortunately, it was not possible to vary the aircraft charge to obtain a set of points for a curve plot, and an accurate relationship between field strength and potential cannot be established. Input impedance to the VTVM was on the order of 10^{10} ohms and sufficient charge leakage took place to decrease the charge voltage by a factor of approximately two. In order to establish a "rough" relationship, two data points along with the zero point were used to plot the linear function appearing in Figure 37. From this plot it appears that the maximum voltage buildup was on the order of 35 KV. A comparison with previous data (Reference 2) establishes this to be a reasonable figure.

ATTITUDE		FLIGHT #1	FLIGHT #2	FLIGHT #3
Hover	25'	+ 30,000 v/m	+ 30,000 v/m	+ 30,000 v/m
	50'	+ 30,000 v/m	+ 30,000 v/m	+ 28,000 v/m
Forward	1000'	+ 16,000 v/m	+ 14,000 v/m	+ 14,000 v/m
Flight	600'	------------	+ 14,000 v/m	+ 14,000 v/m
60 Knots	50'	------------	+ 12,000 v/m	+ 14,000 v/m

NOTE: X10 Mechanical Attenuator Used.

VOLTAGE BUILDUP MEASUREMENT

Attitude	V. T. V. M.*	Sensor
25' Hover	+ 16,000 V.	+ 16,000 v/m
25' Hover	+ 18,500 V.	+ 16,000 v/m
20' Hover	+ 24,000 V.	+ 20,000 v/m

*Hewlett-Packard, Model 410B with 100:1 Probe, Model 459A.

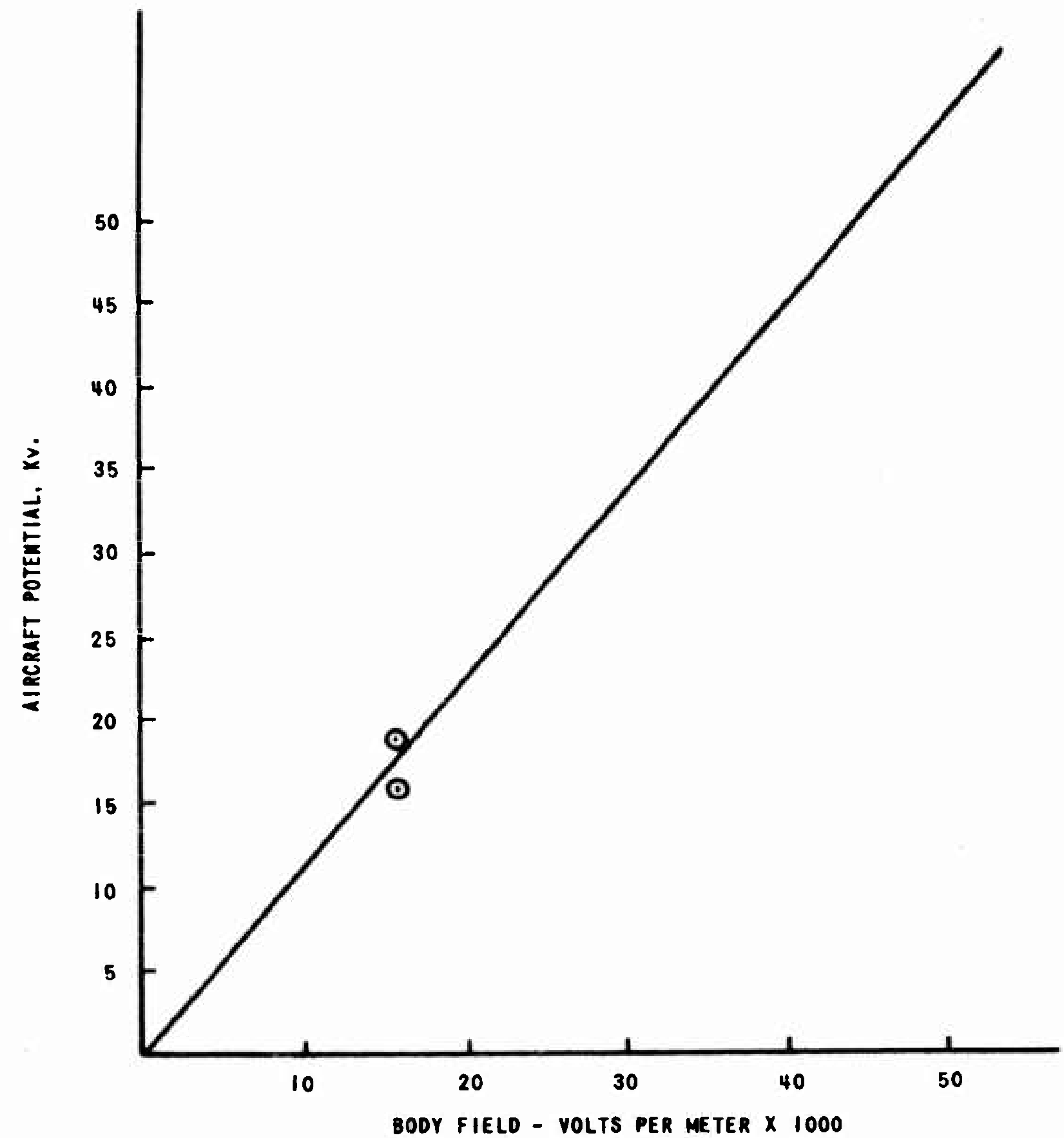


Figure 37 AIRCRAFT H-37 CHARGE VOLTAGE

X APPLICATION TO OTHER AIRCRAFT

The basic system principal and design are valid for other rotary wing aircraft, types YHC-1B, H-34, H-21, and HU-1, however, due to the difference in charging rates, relative capacity and actual aircraft physical configuration, each type should be considered as a separate problem. A comparatively small research effort would be required to determine the following:

1) Aircraft capacity at a 25 foot hover.

2) Relationship of aircraft charge versus aircraft voltage for a particular sensor location.

3) Aircraft charging current.

4) Most effective discharge probe location for a reasonable mounting configuration.

On the basis of information derived, design changes in system parameters can be made to meet the requirements at hand. Determination of aircraft net reduced charge, can be made on the basis of capacity and the charge voltage - field measurement relationship. A change of sensor gain with respect to control circuit triggering levels could then be made to establish proper threshold levels. The high voltage power supply consists of two integral units; primary and control circuitry, and high voltage section. The high voltage section incorporates a quadrupler rectifier and filter network for an output voltage of approximately 24 kilovolts. Through use of a replaceable unit with a doubler or tripler network, the output voltage can be reduced accordingly to meet the aircraft system requirements.

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ACRD(OW), DN	1
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Unclassified Report

A design for a corona discharge system of equipment to maintain automatically a net charge on a helicopter about the zero level was completed and extensive operational tests conducted. An electrostatic (over)

1. Static Charge - Helicopters

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1. Static Charge - Helicopters

field sensor was used to measure the incident field due to the electrostatic charge on the aircraft. An error signal, proportional to the difference between the charge existing on the aircraft and a chosen reference level was used to control operational sequence of a high voltage power supply convected to a corona discharge point. The removal of charge from the aircraft in the form of a discharge current is done in such a manner as to minimize the error within a closed loop system. System equipment was installed in an H-37 helicopter and operational tests conducted in a hot day environment. The aircraft charge was maintained automatically within predetermined limits, corresponding to a net remaining charge voltage of approximately 300 volts with respect to ground.

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